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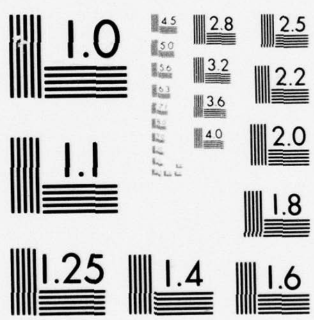
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EVALUATION OF AN AIRBORNE THRUST COMPUTING SYSTEM

VOLUME I SYSTEM PERFORMANCE EVALUATION

Computing Devices Company
Ottawa, Canada

TECHNICAL REPORT ASD-TR-75-2
FINAL REPORT FOR PERIOD JULY 1969 - AUGUST 1973

MAY 1975

Approved for public release; distribution unlimited.

AERONAUTICAL SYSTEMS DIVISION
AIR FORCE SYSTEMS COMMAND
WRIGHT-PATTERSON AIR FORCE BASE, OHIO 45433

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EVALUATION OF AN AIRBORNE THRUST COMPUTING SYSTEM

VOLUME I SYSTEM PERFORMANCE EVALUATION

J.A. Gravelle

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13. ABSTRACT

An in-flight thrust measurement and display system (TMS) for an afterburning turbojet engine with a continuously variable exhaust nozzle has been developed by Computing Devices Company. A special-purpose digital computer calculates engine gross thrust using aerothermodynamic equations requiring ambient static pressure, and three engine tailpipe pressures as inputs. A reference gross thrust is also calculated for a nominal or average engine operating at military power under the prevailing ambient conditions. The display system provides updates of gross thrust and the percentage of the reference gross thrust at the rate of two per second. Flight-rated digital data recording equipment is interfaced with the system to provide a consistency check of the flight data.

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Gross thrust Turbojet Digital computer Airborne Afterburner						

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FOREWORD

This Final Report is submitted in fulfilment of USAF Contract F33657-69-C-0733-P00004, Sequence Number A006 under the designation H036/119/FR/I. The work was administered under the direction of the Propulsion Instruments Branch, Aeronautical Systems Division, Air Force Systems Command, Wright-Patterson Air Force Base, Ohio, by Mr. J.J. Nartker, Project Engineer, (ASD/ENFIP).

This Report covers work performed from July 1969 to August 1973 and was released by the author in October 1973.

The program was conducted by the Research & Technology Division of Computing Devices Company. The development of the aerothermodynamic technology was carried out with Mr. J.R.B. Murphy as the Project Engineer. The specification for the hardware system and the detailed logic and circuit design was conducted with Mr. R. J. Struzina, followed by Mr. H.E. Weissler and subsequently Mr. J.G. Lafeber, as the Project Engineer. The test program was coordinated initially by Mr. J. G. Lafeber and later by Mr. R.I. Alexander.

Dr. D.A.J. Millar, Dr. E.G. Plett and Mr. J.R.B. Murphy developed the gross thrust theory; Dr. G.B. McDonald and Mr. G.B. Mackintosh contributed the engine calibration procedure; and Dr. E.P. Cockshutt of the National Research Council (NRC) provided the reference gross thrust calculation procedure. Special acknowledgements are due Messrs. M.S. Chappell and E.J. Prince of the NRC for their contribution to the engine instrumentation design.

The airborne computing equipment and data acquisition system resulted from the combined efforts of Mrs. S. Brown and Messrs. K.J. Thorstensen-Woll, T.A.C. Rubino, J. Barnhard, G. Foad, J.A. Gravelle and C.H. Henshaw. The reliability and consistent performance of the equipment reflected the quality of the design and its implementation.

The flight test program was conducted at the Aerospace Engineering Test Establishment, Cold Lake, Alberta, Canada and performed from June 1972 to April 1973 under authority of PD 71/81. Capt. E. Morin served as the Project Officer, Major G.M. Smith (USAF) as the Project Pilot and Capt. B.S. Turner as the Instrumentation Engineer.

The ground static evaluation of the installed system was carried out at the National Aeronautical Establishment, Ottawa, Ontario, Canada under the direction of Mr. D. Daw.

Special acknowledgement is due Mr. H. Snowball of the Control Systems Development Branch of the Flight Control Division, Air Force Flight Dynamics Laboratory, Air Force Systems Command, Wright-Patterson Air Force Base, Ohio, for his contribution to the Rutowski Optimum Mission specification.

The author wishes to express his appreciation to the following personnel for their contribution to this Test Report:

Mrs.	S. Brown
Mr.	R.I. Alexander
Mr.	C.H. Henshaw
Mr.	G.B. Mackintosh
Dr.	G.B. McDonald
Captain	J.R.E. Morin
Major	G.M. Smith (USAF)
Mr.	K.J. Thorstensen-Woll
Captain	B.S. Turner

This Test Report is contained in two separately bound volumes.

- . Volume I System Performance Evaluation
- . Volume II System Performance Data (NOTE: VOLUME II CONTAINS ONLY THE RAW DATA).

This Test Report has been reviewed and is approved for publication.

J.J. Nartker
J.J. Nartker, GS-13

John W. Long
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FOR THE COMMANDER

Raymond D. Heinrich
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Chief, Instruments Division
Directorate of Airframe Engineering

EVALUATION OF AN AIRBORNE THRUST COMPUTING SYSTEM

VOLUME I : SYSTEM PERFORMANCE EVALUATION

FLIGHT TEST PERFORMED BY: Aerospace Engineering Test Establishment of the Canadian Armed Forces

FLIGHT TEST AUTHORIZED BY: PD 71/81 - Demonstration and Evaluation of ComDev Thrustmeter, Aug 72

This report contains information and/or recommendations on the performance, design and operation of equipment. None of the recommendations contained herein are to be construed as Canadian Armed Forces endorsement of the products under test. AETE involvement in the test program included installed static tests at NAE Uplands and flight tests at AETE Cold Lake. AETE personnel monitored the ground tests, conducted the flight tests, and participated in data reduction and analysis. The Aerospace Engineering Test Establishment of the Canadian Armed Forces has reviewed Volume I of the report and concurs with the conclusions and recommendations of Volume I .

Reviewed: J E Morin — E. Morin, Capt — AETE Project Officer — Date 3 Oct 73

Gary M Smith — G. Smith, Maj (USAF) — AETE Project Pilot — Date 3 Oct 73

Concurred: L H Keelan — L.H. Keelan, Col — Commander, AETE — Date 3 Oct 73

Approved: A.J. Munroe — A.J. Munroe, Maj — NDHQ/DGAEM/DAES 2-3 — Date 8 May 75

H036/119/FR/I

ABSTRACT

An in-flight thrust measurement and display system (TMS) for an afterburning turbojet engine with a continuously variable exhaust nozzle has been developed by Computing Devices Company. A special-purpose digital computer calculates engine gross thrust using aero-thermodynamic equations requiring ambient static pressure, and three engine tailpipe pressures, and calibration factors determined from ground runs of an engine. A reference gross thrust is also calculated for a nominal or average engine operating at military power under the prevailing ambient conditions. The display system provides updates of gross thrust and the percentage of the reference gross thrust at the rate of two per second. Flight-rated digital data recording equipment is interfaced with the system to provide a consistency check of the flight data.

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LIST OF ABBREVIATIONS

A/B	afterburner
A/C	aircraft
accel	acceleration, as in acceleration loading
A/D	analog to digital
AE	acceptance flight, as in AE-1
AETE	Aerospace Engineering Test Establishment
A/I	anti-icing
AOI	Aircraft Operating Instructions CF-5D, Engineering Order EO 05-205B-1, Canadian Forces, Mar 1970
ASD	Aeronautical Systems Division, USAF
BCD	binary coded decimal
CADC	central air data computer
Cab Pres	cabin pressure
CDP	compressor discharge pressure
CFB	Canadian Forces Base
ComDev	Computing Devices Company
EGT	exhaust gas temperature
EO	Engineering Order used by the Canadian Armed Forces
EPR	engine pressure ratio
F.S.	full scale, defined as 4300 lb_f, the MAX power of a J85-CAN-15 bare engine at sea level standard day conditions
FT	flight trial, as in FT-01
flt	flight
GE	General Electric
LFR	fuel remaining, left system

LIST OF ABBREVIATIONS (CON'T)

MAX	maximum afterburner power setting
MIL	military power setting
MTBF	mean time between failures
NAE	National Aeronautical Establishment (NRC)
Norair	Northrop Corporation, Norair Division
NPI	nozzle position indicator
NRC	National Research Council
PFRT	pre-flight rating test
PD	project directive
PLA	power lever angle
PT	pressure transducer
QTP	qualified test pilot
RFR	fuel remaining, right system
RPM	rpm, usually as % maximum rated rpm (16,500) for J85-CAN-15 engine
TMS	Thrust Measuring System
USAF	United States Air Force
VEN	variable exhaust nozzle

LIST OF SYMBOLS

A	area
C_D	drag coefficient
C_{DL}	drag coefficient due to lift
ΔC_D	incremental drag coefficient
C_{5-6}, C_{6-7}	engine calibration constants
γ	ratio of specific heats
D	drag force
σ	ratio of ambient static to sea level standard static pressures
E	engine calibration constant; also total energy according to the context
E_h	specific energy
F	thrust or force
F_G	gross thrust
F_R	reference gross thrust
F_N	net thrust
F_{RDG}	ram drag
ΔF_G	incremental gross thrust
ΔF_N	incremental net thrust
$f(\beta)$	nozzle performance function
G-loading	acceleration loading equal to that produced by the earth's gravitational acceleration is referred to as a 1-G loading
g	acceleration due to gravity at a point
g_o	gravitational constant $\left(32.174 \dfrac{\text{ft lb}_m}{\text{lb}_f \text{ sec}^2}\right)$

Hp	pressure altitude
h	geometric altitude
$h(\alpha)$	engine performance function
θ	ratio of ambient static to sea level standard static temperatures
λ	lag factor for % reference thrust indicator
M	Mach number
$\dot{m}$	mass flow rate
N	engine rpm
P	pressure
ΔP	differential pressure $\equiv P_{T5} - P_{S6}$
ΔP_S	differential pressure $\equiv P_{S6} - P_{S7}$
S	wing area
σ	standard deviation
T	temperature
T_M	net propulsive thrust
T_{5X}	aerodynamic mean total temperature at Station 5
t	time
V	velocity
$\dot{W}$, W_{fm}	fuel flow
W	aircraft weight

Subscripts

A/B	afterburner
a	air; also a P_{S7} location
b	a P_{S7} location

LIST OF SYMBOLS (CONT'D)

ej	ejector
f	fuel
G	gross
H	harness, as in T_{5H}
i	indicated
MAX	maximum
MIN	minimum
R	reference
s	static or standard depending on context
T	total
0	ambient
2	compressor inlet location
3	compressor discharge location
5	turbine exit location
6	a tailpipe location (see Figure 35)
7	nozzle entrance location
8	nozzle throat location

LIST OF DEFINITIONS

a (or) b; identification of the P_{S7} probe locations.

adjustable constant; calibration constant defined as a result of bare engine running and a mathematical technique. TMS hardware was adjustable until such time as the constants could be defined.

BCD; a number consisting of a group of four figures that represent, but not necessarily equal arithmetically, a figure in an associated decimal number; e.g. the BCD number representing the figure 6 is 110.

bit; single character in a binary number. (<u>b</u>inary digi<u>t</u>)

chop; a rapid power lever movement to reduce engine power.

engine auxiliary takeoff door, or takeoff door, or engine auxiliary air inlet door; a louvered door located on the inlet duct and used to furnish additional intake air to the engine during takeoff and landing.

full scale (F.S.); the maximum bare engine gross thrust at sea level on a standard day. (Rating is 4300 lb_f according to the AOI).

gross thrust; thrust developed within an engine as a result of gas flow and heat addition between the exhaust gas turbine exit and the nozzle exit plane. See also paragraph 1.4.3.

indicator; digital and pointer display used as an aircraft cockpit instrument to display engine gross and percent reference thrust.

interface; an electronic hardware boundary between systems or parts of a system.

maximum A/B, or MAX A/B; maximum power available when the engine is operated in the afterburning mode.

military power, or MIL power; a throttle setting marked by a throttle position detent. MIL power is the maximum power available without afterburning.

percent reference thrust; the percent of nominal MIL power thrust for an average engine under the prevailing flight conditions.

reference thrust; reference gross thrust developed at MIL power by an average engine under the prevailing flight conditions.

LIST OF DEFINITIONS (CON'T)

slam; a rapid increase in engine power achieved by movement of the power lever.

standard deviation; a statistical description of the dispersion of values about the mean value. As used in this Volume, the standard deviation, σ, is computed as the square root of the sum of the squares of the deviation between TMS data and test stand data divided by the number of data points less one.

test life; length of time accumulated without failure, as applied to indicating the number of hours of ground running tests and flight tests the pressure probes accumulated while in a serviceable state.

tracking; the ability of the TMS to follow variations of the test engine thrust. Engine thrust variations are the result of operating the engine power lever.

transducer; a hardware item used to sense pneumatic pressure and output a correlated electrical signal.

DISPOSITION OF TEST EQUIPMENT

Hardware used in the thrust measuring system trials either has been, or will be disposed of according to the following schedule.

ITEM	QUANTITY	DESCRIPTION	DISPOSITION
1	3	Indicator, thrustmeter	1
2	1	Thrust Computer	1
3	2	Central Air Data Computer	2
4	2	Temperature Probe,ambient air, Rosemount	2
5	1	Exhaust Gas Temperature Indicator	2
6	2	Transducer, NAE thrust stand	2
7	1	Tray, mounting, thrust computer	1
8	1	Tray, mounting, data recorder	1
9	1	Power Supply, data recorder	4
10	1	Photopanel c/w instrumentation less time display (Item 11) and thrustmeter indicator (Item 1)	2
11	1	Time Display	1
12	3	Transducer, SE Laboratories, pressure	3
13	1	Recorder, Incre-data magnetic tape, digital	4
14	6	Transducer, Conrac, pressure	1

DISPOSITION ACTIONS

(1) These items are the property of the customer and are to be stored at ComDev pending disposal orders from the customer.

(2) These items are the property of the Canadian Armed Forces and are to be returned to the Aerospace Engineering Test Establishment, CFB Cold Lake, Alberta. The CADC will be demodified prior to being returned.

(3) These items were originally loaned to ComDev by the manufacturer. They have since been presented to ComDev for the purpose of long term testing in the ComDev laboratories.

(4) This item is the property of ComDev and will be stored until required for future projects.

SUMMARY

This Volume describes the thrust measuring system, test method and test equipment. Testing included bare engine tests, installed static thrust trials and airborne flight testing. Test observations and results are reported. It is recommended that:

(a) A refined and flight tested TMS based on the concept proven and reported herein be used both as a cockpit indicator of engine performance and as a maintenance tool,

(b) A miniature computer for a TMS be designed for fighter type aircraft,

(c) Pressure probe design be improved for TMS use in service engines,

(d) Some further bare engine testing should be conducted to demonstrate TMS response to artificially induced engine malfunctions,

(e) Some further analytical work be undertaken to develop a system for turbofan engines,

(f) An investigation be made into the fact that the reference thrust indicator did not always indicate 100% reference thrust when the engine was operated at military power at altitude; and

(g) Some minor improvements to the indicator display should be considered.

SECTION I

INTRODUCTION

1.1 BACKGROUND

1.1.1 Aircraft engine design technology has been advancing rapidly for many years. As a result, a variety of relatively sophisticated power plants have been developed both for commercial and military use. The problem of measuring in-flight engine thrust has long been recognized and many thrust measuring systems have been proposed. Prior to the system reported here, an entirely satisfactory thrust measuring device for afterburning engines with variable nozzles had not been developed.

1.1.2 Computing Devices Company (ComDev) began researching the thrust measuring problem in 1967 following discussions between ComDev and the (Canadian) National Research Council (NRC). ComDev received a contract in 1969 which was funded jointly by the Government of Canada and the USAF Propulsion Instruments Branch, Aeronautical Systems Division (ASD).

1.1.3 ComDev developed the theory needed in order to design and build a thrust measuring system for afterburning, variable nozzle turbojet engines. The system was constrained to operate without fuel flow, engine temperature, nozzle total pressure or nozzle area intelligence. An advanced development model TMS consisting of three sensors, computer and two indicators was designed, built, ground tested and flight tested. Ground testing included bare engine tests conducted at the Engine Laboratory of NRC and installed engine tests at the National Aeronautical Establishment (NAE) of NRC. Flight testing was accomplished by installing the system in a CF-5D aircraft. Flight tests were conducted by the Aerospace Engineering Test Establishment (AETE) of the Canadian Armed Forces.

1.2 OBJECTIVES

1.2.1 A gross thrust measuring system (TMS) has been designed and built based upon aerothermodynamic theory developed by ComDev. The TMS uses engine pressure data and ambient pressure data in solving a mathematical equation which produces thrust data. Since the TMS made use of certain previously unproven theories and innovations, it was most important that the concept be flight tested and evaluated. Therefore, a test program with the following objectives was undertaken:

(a) Obtain calibration data. TMS theory indicated that a system could be designed to operate within the specified constraints provided an engine of the type for which the system was to be built could be used to obtain calibration constants. The calibration constants would become an integral part of the thrust equation and would be valid throughout the flight test envelope. A J85-CAN-15 engine was selected for this purpose.

(b) Debug and test the TMS. A TMS computer and two indicators were designed and built. Bare engine running tests were required in order to debug the system and to demonstrate its responses to actual dynamic engine operations. Test cell running was also required in order to evaluate the TMS accuracy.

(c) Prove flight safety of engine hardware. A pre-flight rating test (PFRT) was required to prove the engine hardware prior to actual flight testing.

(d) Demonstrate TMS installed operation. The J85-CAN-15 is used in the Canadian Armed Forces CF-5D aircraft. Installed static trials were required in order to demonstrate the system performance in the installed configuration.

(e) Demonstrate airborne operation. Static trials cannot demonstrate the TMS operation at high altitude or G-loading or other normal flight conditions. Although comparative thrust data are not available during flight, it was necessary to test and evaluate the TMS throughout the operating envelope of the test aircraft.

(f) Obtain qualitative and quantitative data. Flight testing was required in order to provide qualified test pilot qualitative evaluations as well as quantitative performance data. The latter data were required in order to demonstrate the potential usefulness of a TMS in an operational environment.

(g) Evaluate repeatability and reliability. System repeatability and hardware reliability data were required in order to evaluate the system performance.

1.3 SCOPE

1.3.1 This volume presents a description of the TMS, the equipment to test the system, test procedures and test results. A number of operational uses for the TMS have been examined and reported in detail. Conclusions drawn from the TMS testing and a number of recommendations are made.

1.4 GENERAL TECHNICAL INFORMATION

1.4.1 Testing agencies use gross and net thrust data in evaluating engine and airframe performance. At the present time, working without a thrust measuring system, engine performance is tested in a static engine thrust cell where many variants such as fuel fl[illegible], engine pressure ratio, temperatures and pressures are measured. In-flight thrust is frequently estimated by measuring these variants and a computer is used to resolve thrust estimates. Computer programs for this purpose are usually prepared by the engine manufacturer. This system depends upon accurate, continuous measurement and recording of many variables. The method may be degraded by the assumption that static, ground level engine test data can be extrapolated throughout the aircraft operational envelope. Also, in-flight thrust data are not available to the pilot. The ComDev TMS provides continuous data which are dependent only upon relatively easily obtained engine pressures and ambient data from the aircraft central air data computer (CADC). This system should be accurate for all engine operations for which the TMS gross thrust equation and calibration are valid.

1.4.2 A percent reference thrust indicator provides the pilot with a means of assessing engine performance by comparing actual gross thrust to reference gross thrust. Reference gross thrust is simply a computed value which should be expected for the test day pressure, temperature and flight Mach number. Naturally this computation depends upon an accurate definition equation for the reference engine. Provided this is practical, the percent reference thrust indicator could warn a pilot of an engine fault which could be caused by icing, foreign object ingestion or perhaps a faulty fuel or nozzle control.

1.4.3 Gross thrust is defined as the exit momentum (aligned with the center-line of the nozzle) plus a pressure-area term to account for incomplete expansion to the prevailing ambient static pressure at the exhaust plane of the primary nozzle.

1.4.4 The ComDev TMS computer was designed and built in order to prove the TMS concept. Very little effort was made to miniaturize the computer. It is recognized that a TMS for fighter aircraft must be of considerably smaller dimensions. It is within the state of the art of present day computers to build a much smaller TMS computer.

SECTION II

DESCRIPTION OF TEST EQUIPMENT

2.1 TEST ENGINE

2.1.1 The test engine was a General Electric J85-CAN-15, eight stage, axial flow, turbojet engine equipped with an afterburner, see Figure 1. The sea level standard day, static rated bare engine thrust is 4300 pounds at maximum afterburner (MAX) power and 2925 pounds at military (MIL) power.

2.1.2 Engine air enters the aircraft through side fuselage inlet ducts and is directed into the compressor sections by variable inlet guide vanes. During aircraft takeoff, additional air is provided by engine auxiliary takeoff doors on the side of each fuselage air inlet duct, see Figure 2. Automatically controlled inlet guide vanes and compressor air bleed valves reduce the possibility of a compressor stall. Compressor bleed air is used to supply various pneumatic systems.

2.1.3 The engine has a single-rotor eight-stage compressor coupled directly to a two-stage turbine. Exhaust gases from the combustor section pass through the turbine section and are discharged through a variable area nozzle. The variable exhaust nozzle (VEN) is electrohydraulically actuated. Below 95% RPM, the VEN is controlled by throttle setting. At MIL and afterburning throttle settings, the VEN is temperature controlled to provide optimum thrust efficiency without exceeding exhaust gas temperature limits.

2.1.4 For TMS flight testing purposes, a standard J85-CAN-15 engine, serial number 8476, was modified only as required in order to install total and static pressure probes. The static pressure ports do not disturb the engine gas flow as they are in effect simply 0.031 inch diameter holes drilled in afterburner liner fittings. Figures 3 and 4 present diagrams of the P_{S6} and P_{S7} static pressure probes used in the flight tests. Four total pressure probes, as shown in Figure 5, were installed on the diffuser struts. These 0.188 inch diameter probes will produce a negligible effect on the gas flow. Probe locations are shown in Figure 1.

2.2 TEST AIRCRAFT

2.2.1 The CF-5D aircraft is a Canadian Armed Forces, two-place, high performance tactical fighter and trainer. This aircraft has twin J85-CAN-15 engines and is capable of supersonic flight. The rear cockpit is equipped with dual controls for pilot training. The fuselage is an area rule shape. The wing, horizontal stabilizer, and vertical stabilizer leading edges are moderately swept back. Each wing incorporates both leading and trailing edge flaps for increased lift and improved low-speed handling characteristics. A speed brake is located at the lower mid-center of the fuselage.

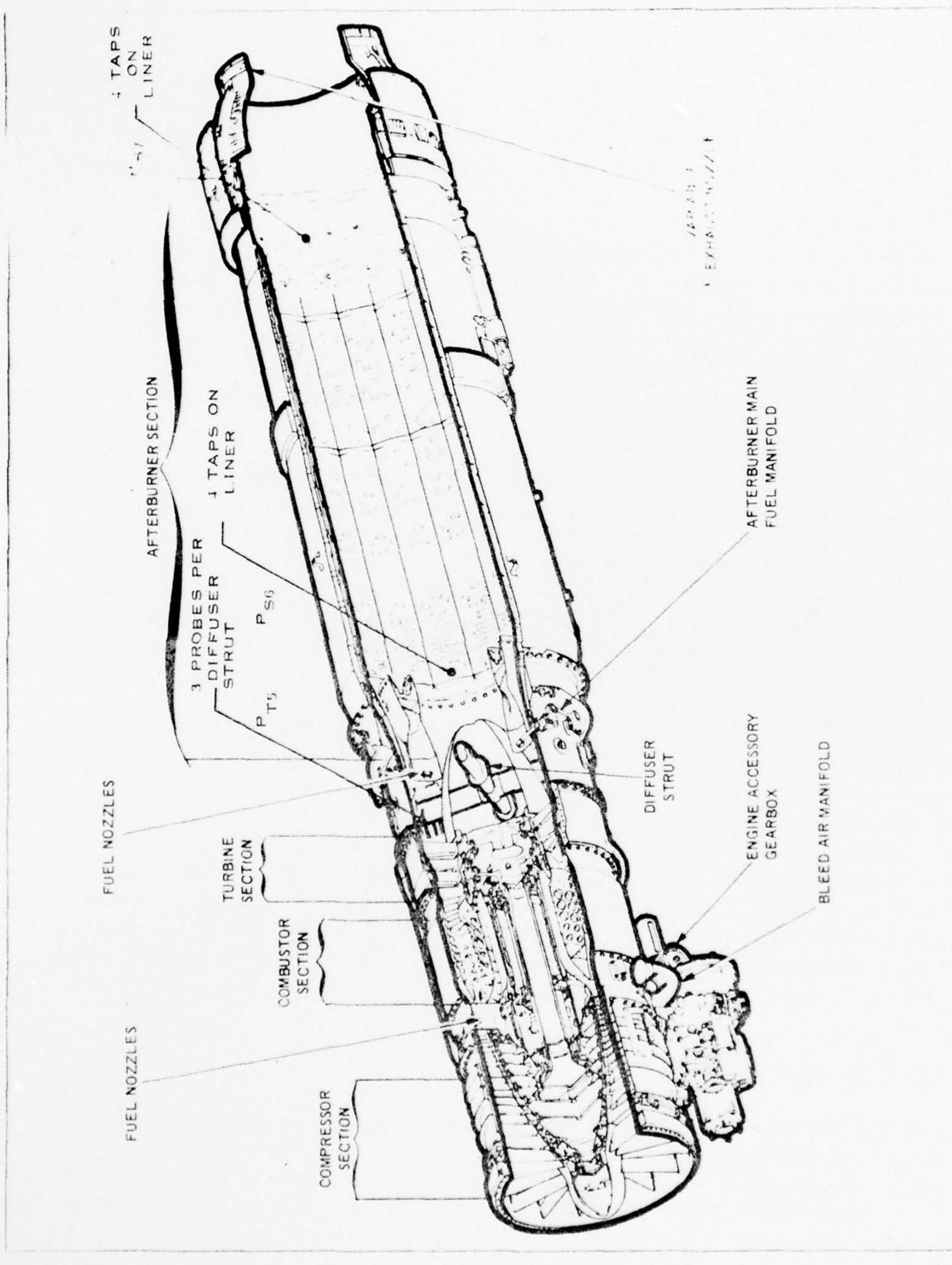


Figure 1 J85-CAN-15 Turbojet Engine

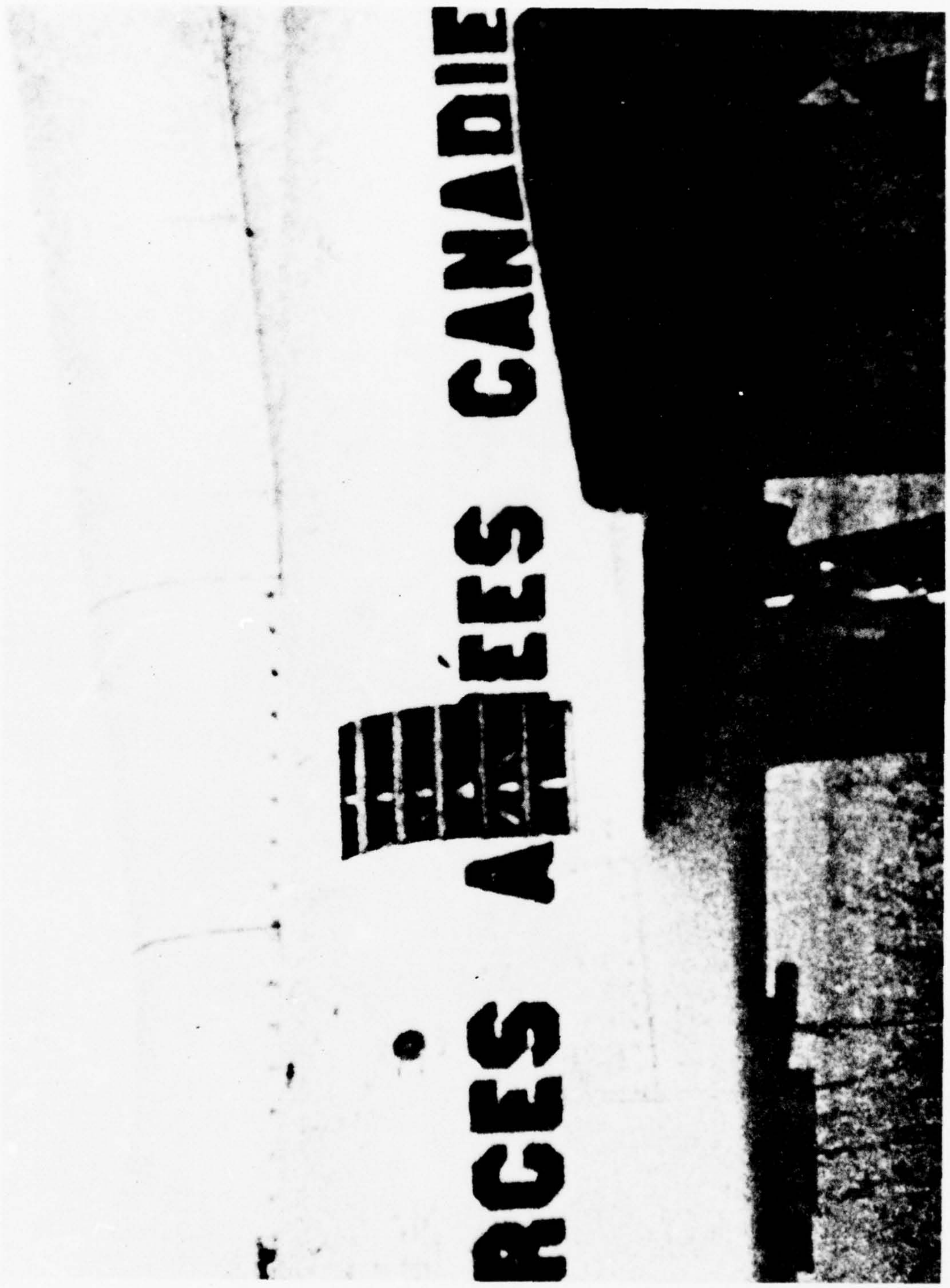


Figure 2: CF-5D Aircraft Engine Auxiliary Takeoff Door in the Open Position

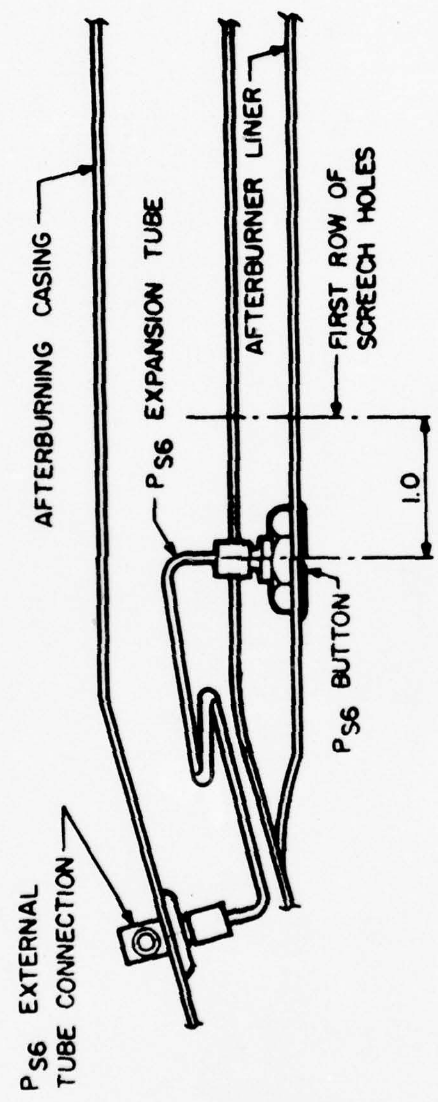


Figure 3: P_{S6} Static Probe for J85-CAN-15 Engine

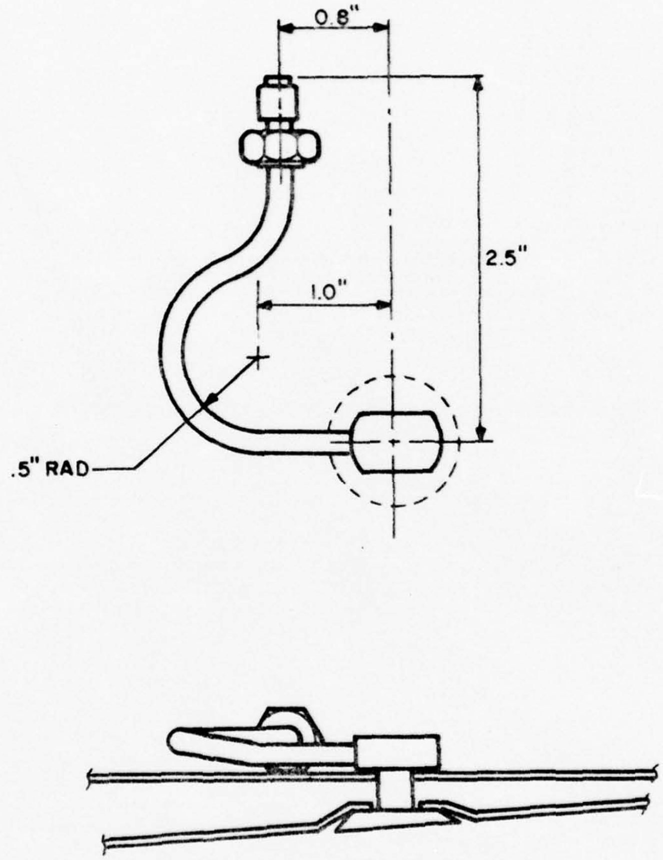


Figure 4: P_{S7} Static Pressure Probe for J85-CAN-15 Engine

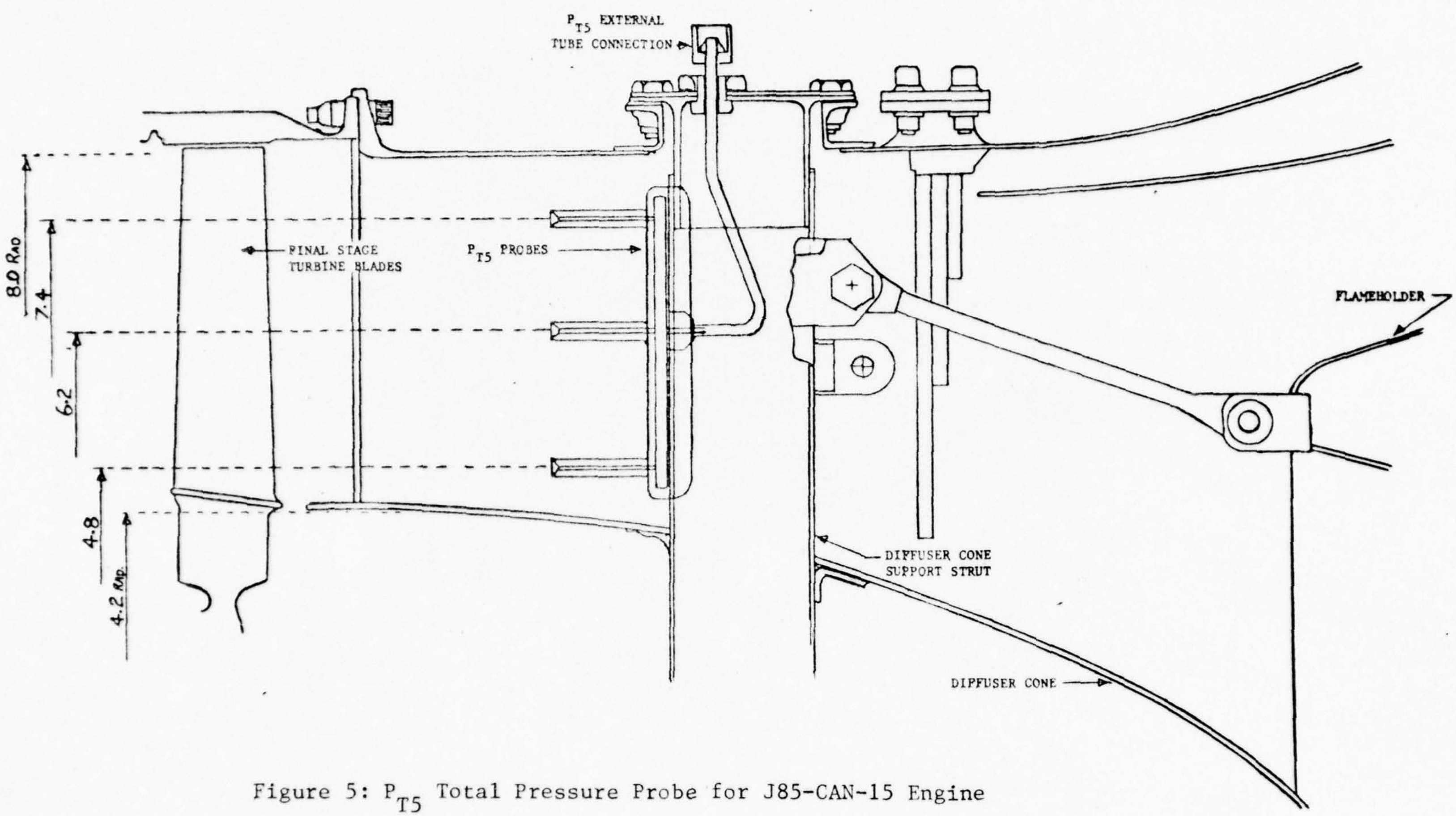


Figure 5: P_{T5} Total Pressure Probe for J85-CAN-15 Engine

2.2.2 Thrust is supplied to the CF-5D aircraft by two axial flow, turbojet engines equipped with afterburners. Each engine is provided with an inlet duct and a louvered auxiliary air inlet engine door on each side of the fuselage.

2.2.3 In the TMS flight trials, CF-5D serial number 116801, was used in the clean plus wingtip tank configuration. The test pilot used the forward cockpit. A data recorder, TMS computer and a photopanel were installed in the rear cockpit. The ejection seat and control column were removed from the aft cockpit. The aircraft plus recorder and TMS, less pilot and fuel weighed 9730 lbs. The aircraft, as above, plus fuel and aircrew weighed 14345 lbs. Figure 6 is a photograph of the test aircraft. The TMS computer and data recorder installations are shown in Figure 7.

2.3 Thrust Measuring System Components

2.3.1 Thrust Computer System

2.3.1.1 The thrust computing system, shown in Figure 7, consists of a special purpose digital computer, three pressure transducers and two indicators. Additional inputs are required from the Central Air Data Computer (CADC). The ComDev gross thrust calculation method is described in Appendix I. The reference gross thrust calculation method is explained in Appendix II. The thrust indicator displays engine gross thrust in pounds and the ratio, expressed as a percent, of the actual engine gross thrust to the engine installed nominal value, military power, gross thrust for the prevailing flight conditions. Three pressure transducers are used to sense tailpipe pressures and provide data for the computer. Engine pressure probes are mounted at engine stations 5,6 and 7 as depicted in Figure 8. The computer solves thrust equations and outputs electronic data which drives the indicators.

2.3.2 Transducers

2.3.2.1 Three engine pressure transducers are required in order to sense total and static pressures within the afterburner section of the engine. Two differential transducers were used to measure pressure differences between the total pressure at station 5 and the static pressure at station 6, ΔP, and between the static pressures at station 6 and 7, ΔP_S. An absolute pressure transducer was used to measure the static pressure at station 7, P_{S7}. The transducers were installed in the aircraft rear cockpit where they would be exposed to a relatively moderate environment.

2.3.2.2 Variable reluctance transducers were provided for the flight trials by S.E. Laboratories, Ltd., Feltham, UK. Static calibrations indicated that the transducers were very accurate and stable. Calibration data are presented in Volume III and serial numbers are listed in Appendix XI.

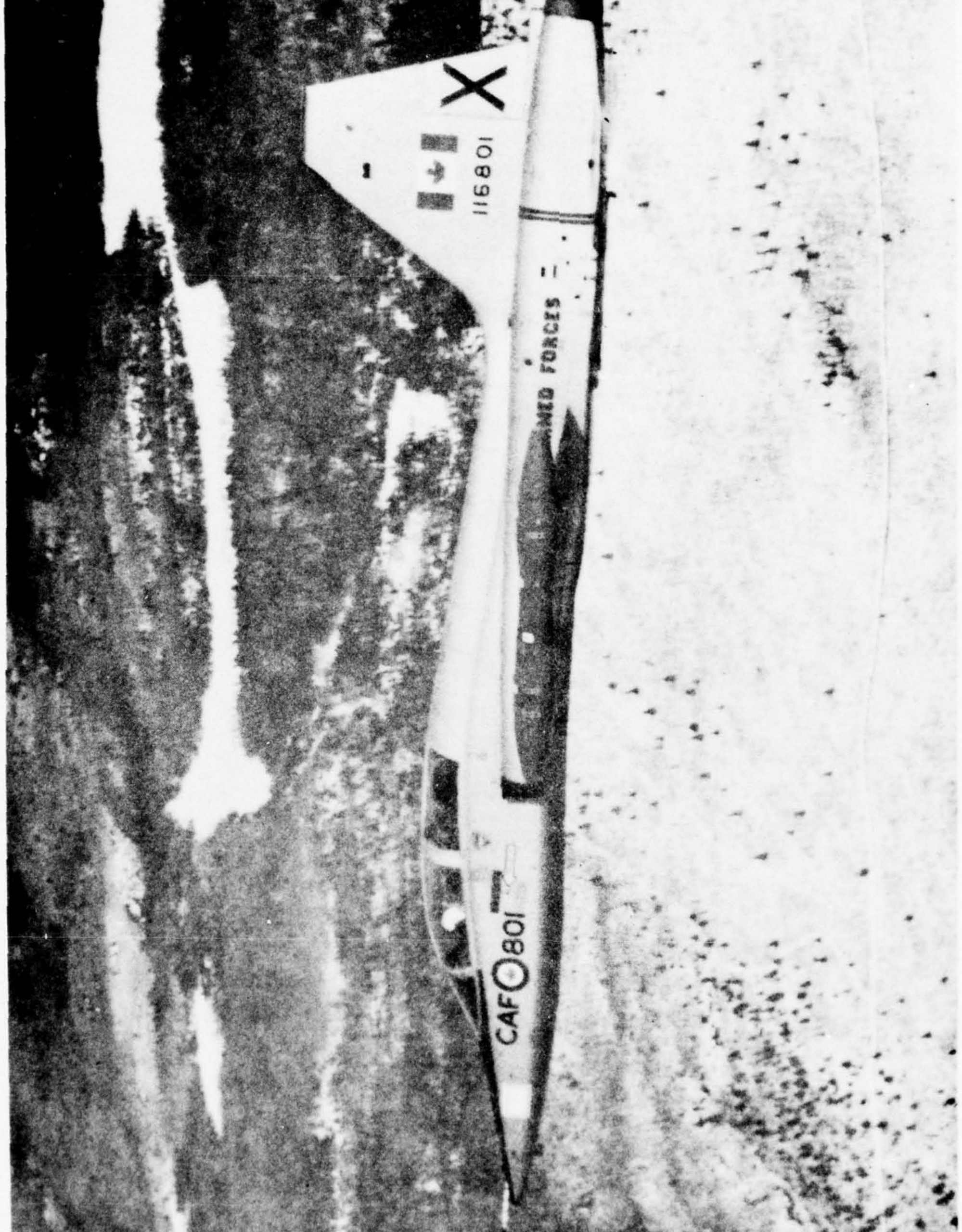


Figure 6: CF-5D Aircraft Used for the TMS Flight Test

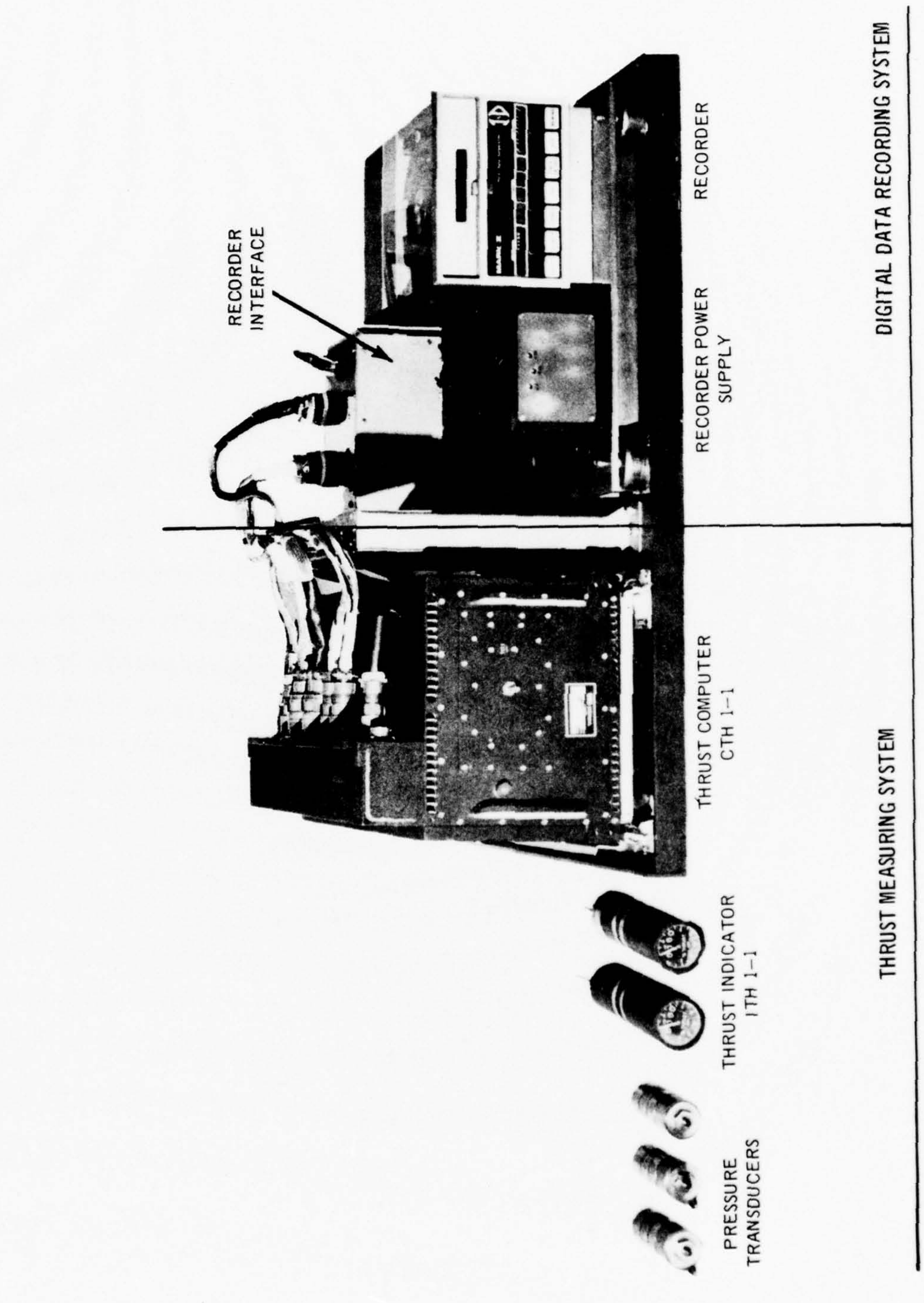


Figure 7: Thrust Measuring System and Magnetic Tape Recorder

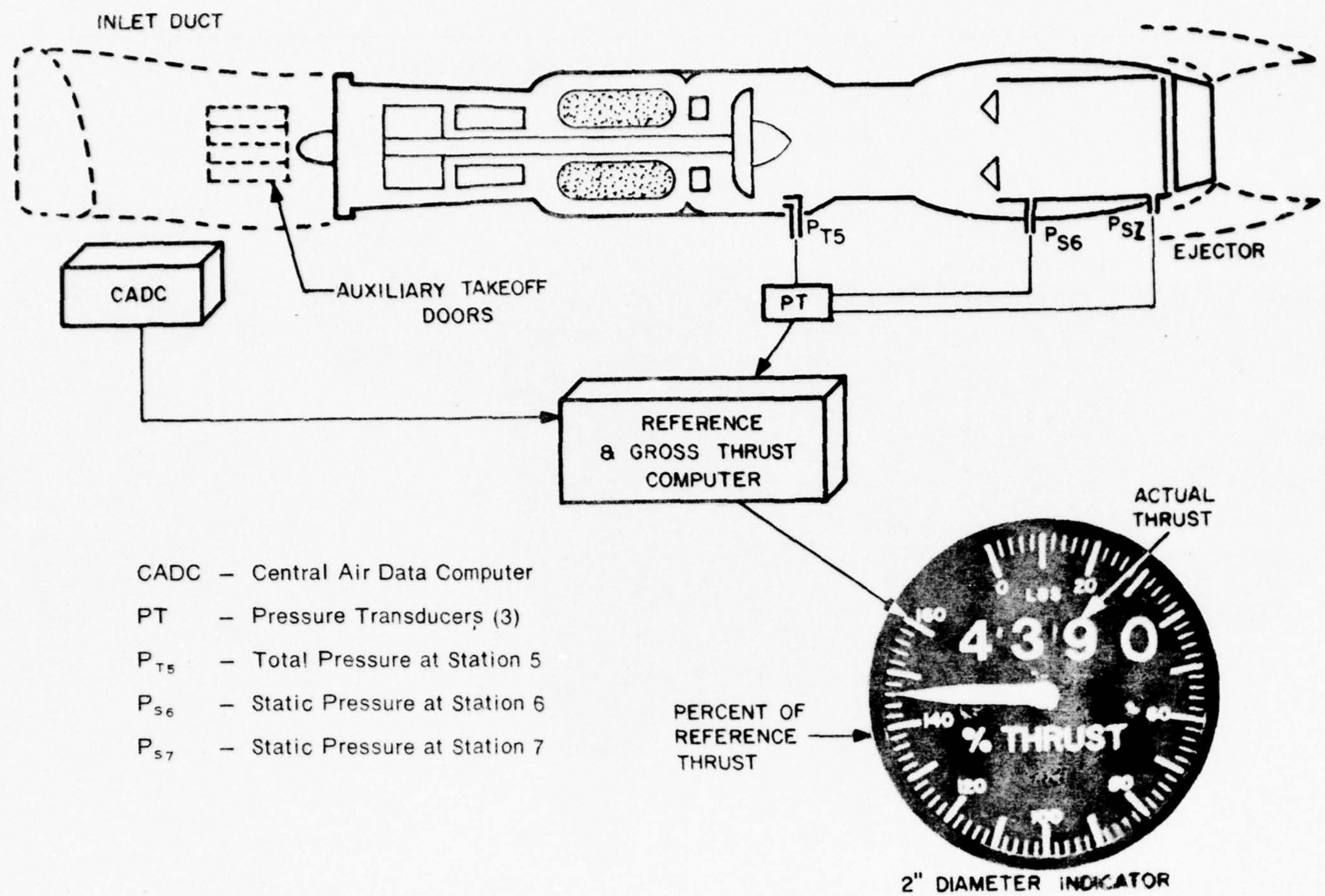


Figure 8: Operation of Thrust Measuring System

2.3.2.3 Three types of Conrac Corporation, Duarte, California, Model 4715H transducers were obtained and tested in the static thrust stand trials. They were removed from the TMS prior to the flight trials. Calibration data are presented in Volume III and serial numbers in Appendix XI.

2.3.3 Special Purpose Computer

2.3.3.1 The computer for the TMS is a special purpose, 21-bit hardware floating point machine which was designed and built by ComDev especially for the TMS. The computer contains an analog multiplexer to select in turn the seven input analog signals and to route these to an analog to digital converter (A/D). An A/D converter is used to transform analog signals to their equivalent digital representation for the central processor. The central processor performs arithmetic operations as required in solving equations for gross and percent reference thrusts. Processor outputs are digital signals which drive the indicators. A flow diagram of the arithmetic units is shown in Figure 9.

2.3.4 Thrust Indicator

2.3.4.1 A two-inch diameter indicator is used to display gross and percent reference thrust as depicted in Figure 8. The digital gross thrust display contains a binary to binary coded decimal (BCD) encoder which interfaces the computer to a set of digital drums. Four digits are used but the units digit is a fixed zero. Percent reference thrust is a pointer display which is driven by a stepper motor. A shaft encoding system is used to control the motor operation. The TMS computer was used to drive two indicators, one in the forward cockpit and one in the photopanel.

2.3.5 Modified CADC

2.3.5.1 The CF-5D aircraft is equipped with a central air data computer (CADC) which provides data to an AVU-9/A airspeed and Mach number indicator and an AAU-19/A altimeter. Ambient air probes are installed on the aircraft and the CADC senses probe pressures and converts them into electrical signals for the indicators. The CADC corrects the sensed data for position errors and, therefore, the CADC output data are highly ideal as input to the TMS computer. A standard CADC was modified by Airesearch Manufacturing Company, Torrace, California, in order to provide output lines for the TMS as follows:

(a) Mach number output is a voltage linearly proportional to Mach number between the range 0.17M to 1.7M.

(b) Static pressure, P_{SO}, is a voltage linearly proportional to ambient static pressure.

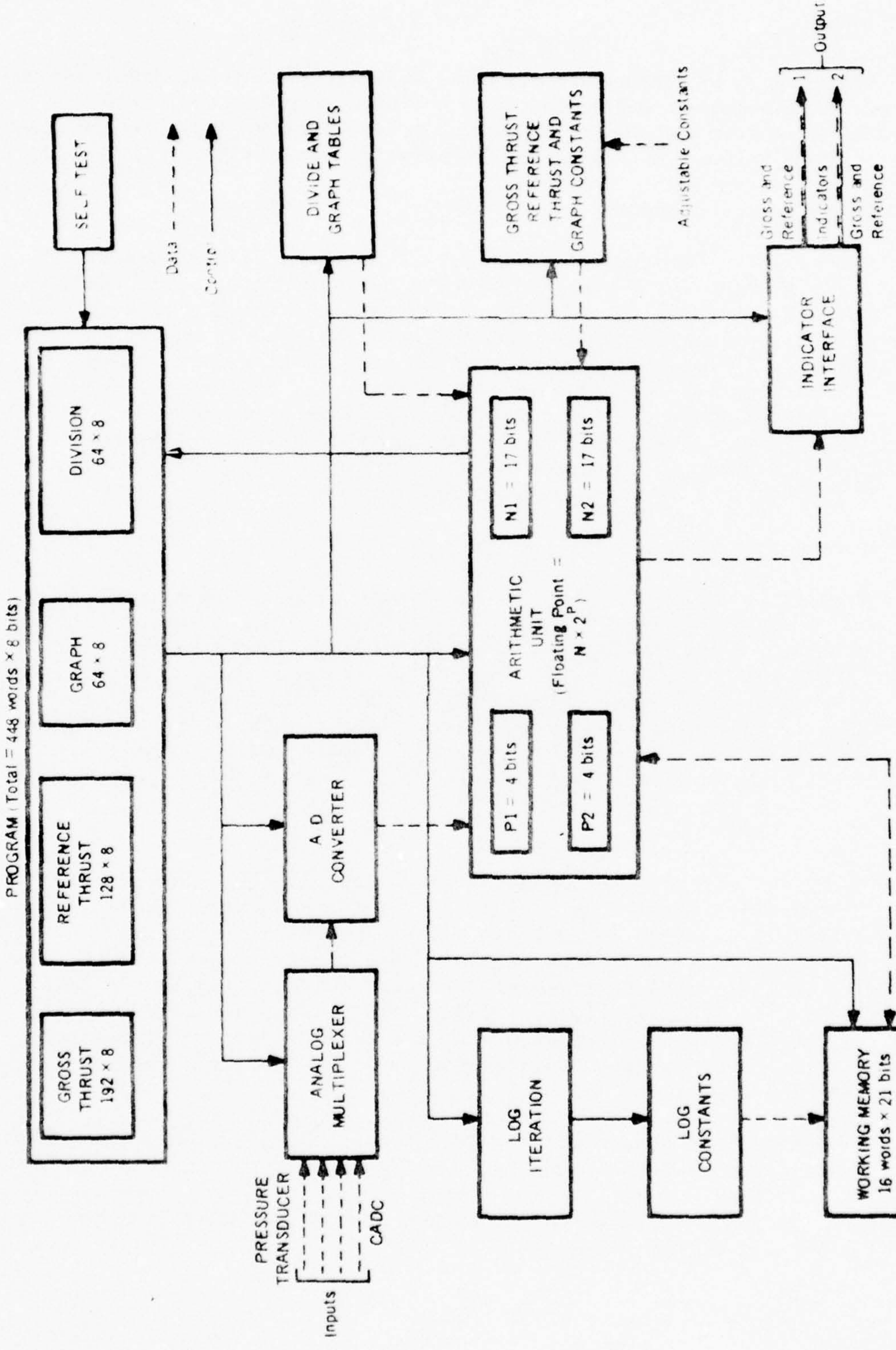


Figure 9: Thrust Computer Arithmetic Units

(c) Temperature, T_{TO}, is a voltage linearly proportional to the square root of the indicated total ambient temperature $\sqrt{T_{Ti}}$.

2.3.5.2 Modifications were designed such that demodification was easy and normal CADC operation was possible.

2.3.6 Temperature Probe and Pre-Amplifier

2.3.6.1 Ambient air total temperature data were required by the reference thrust equation. A data source was not readily available in the standard CF-5D CADC. Therefore, a Rosemount temperature probe was installed on the port side of the aircraft nose section. Airesearch modified the flight test CADC to accept temperature probe signals and output a voltage proportional to $\sqrt{T_{Ti}}$. Since the CADC output voltage range varied by only 0.1 volt for a temperature range from 360° to 740°R, a pre-amplifier was required to increase this voltage prior to its use in the TMS computer.

2.4 DATA ACQUISITION SYSTEM

2.4.1 Four distinct data acquisiton systems were employed in monitoring the thrust measuring system and aircraft variants. Two systems depended upon observations made by the qualified test pilot (QTP), namely pilot observations and taped voice recordings. Two other systems operated independently from each other and except for on-off actuation, were independent of the pilot. These were a multiple input digital tape recording system, shown in Figure 7, and a photopanel shown in Figure 10. The test pilot could command the operation of a cinecamera and obtain a photographic record of the photopanel. A detailed description of the data acquisition systems may be found in Appendix III of this Volume.

2.4.2 Digital Recorder. An Incre-data Corporation Mark IIA, 7-track digital, magnetic tape data recorder was installed in the rear cockpit of the test aircraft as shown in Figure11 and used to record TMS, engine and flight data. Recordings were made at one-half second intervals throughout every flight and most of the installed static thrust trials. A list of recorded variants is presented in Appendix III. Tapes produced by the recorder were compatible with data processing computers at ComDev and with IBM computers in Edmonton and Ottawa. An electronic interface was designed and built by ComDev in order to couple the various TMS data sources to the digital recorder. A second interface between the aircraft signals and the recorder was built by AETE. Input to the recorder was in the form of a voltage in the range 0 to +5 volts. The recorder converted voltages to 12-bit binary data in the range 0 to 4095. Bench test calibrations were made in order to provide the data reduction computer program with an arithmetic means of converting binary data to engineering system equivalent data.

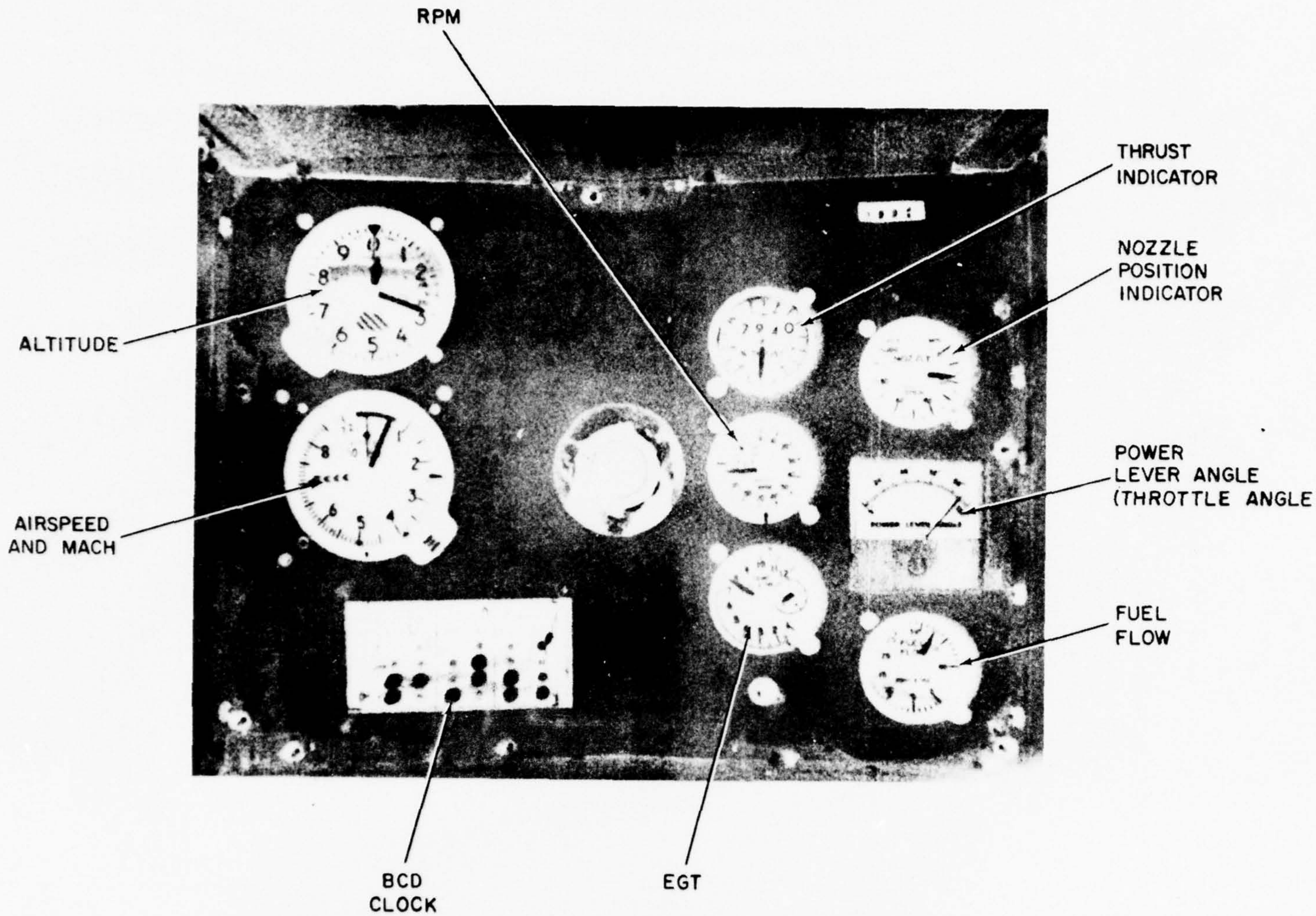


Figure 10: Photopanel Used in Testing the TMS

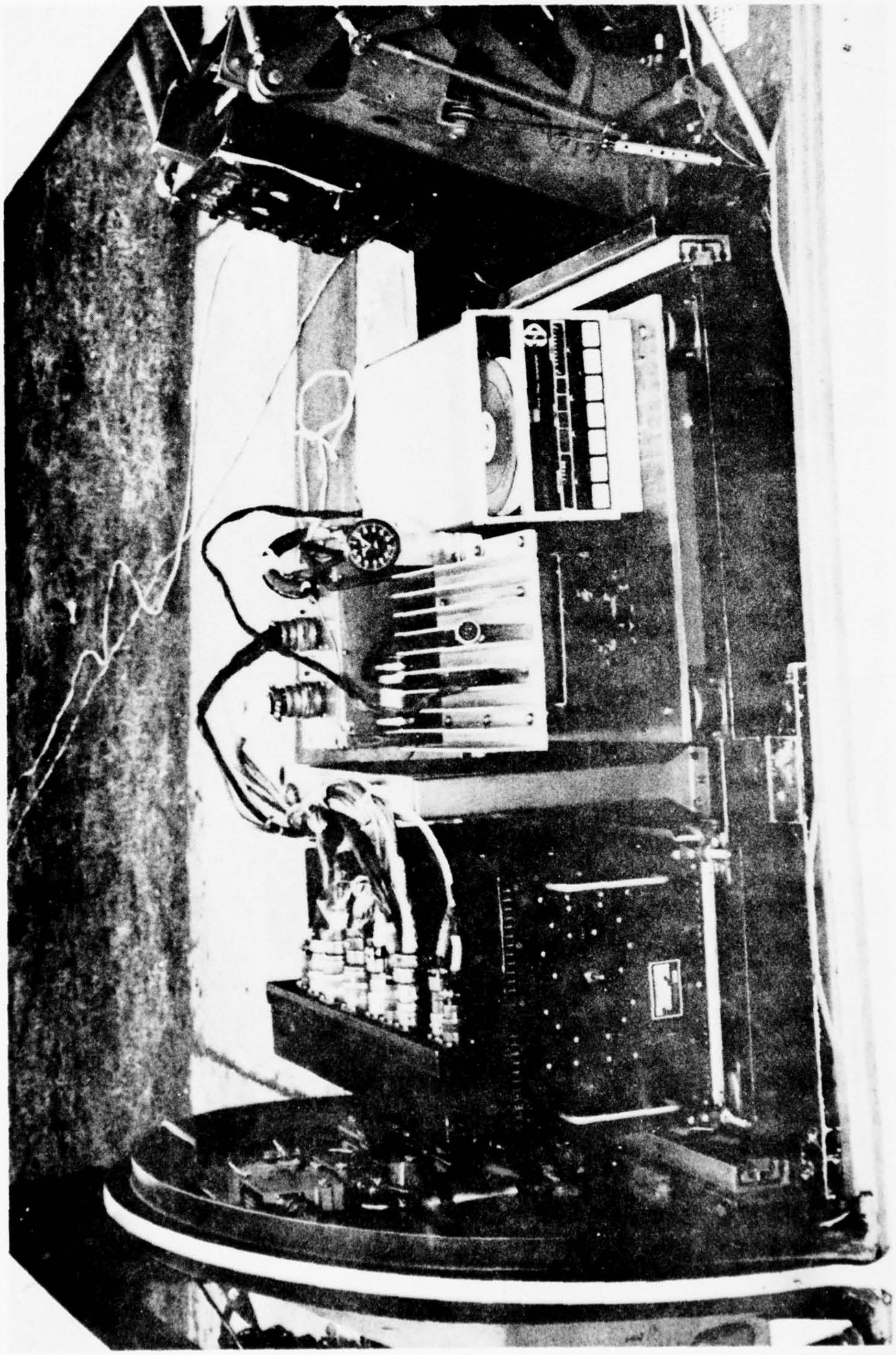


Figure 11: Magnetic Tape Data Recorder Installed in the Test Aircraft

2.4.3 Photopanel. A special AETE built panel of instruments, as shown in Figure 10, was mounted in the test aircraft rear cockpit. The panel contained a thrust indicator as well as a standard set of CF-5D engine instruments, an altimeter, and an airspeed and Mach number indicator. A binary coded clock on the panel was synchronized with the digital tape recorder such that photopanel and recorded data could be correlated. A 16mm camera was set to photograph the panel on pilot command.

2.4.4 Pilot's Voice Recording. The pilot maintained radio contact with engineers in a ground station. Pilot transmissions describing his actions and selected instrument readings were recorded at the ground station for post-flight analysis purposes.

2.4.5 Pilot Observations. Flight missions were pre-arranged and recorded on the QTP's knee pad. In-flight notes were also made on the pad report form. Pilot comments were recorded at debriefings following each flight.

2.4.6 Aircraft Variants. A number of variants pertaining to the operation of the aircraft were recorded by the digital tape recorder. These data were used in analysing engine performance and flight conditions which could be correlated to the performance of the TMS. A brief description of these variables follows. An interface was designed and built by AETE to allow digital data recording of these variants.

2.4.6.1 Fuel Flow. Special fuel flow turbines and transmitters, (Foxboro type 3/4-81T3C1 for the afterburner and type 3/4-81T3A1 in the main fuel line) were installed in the test aircraft. The turbines were calibrated using Calibrating Fluid MIL-C-7024B Type II. Turbine data output is a frequency which was calibrated to pounds per hour. Frequency data were converted to voltage data by the interface.

2.4.6.2 T_{T5H}. Engine total temperature at Station 5 (exhaust gas temperature) was obtained from the harness, T_{T5H}, circuit in the aircraft engine.

2.4.6.3 Psej. Ejector static pressure data were obtained by installing four static pressure ports in the ejector wall. The ports were spaced about the circumference of the ejector and approximately at the minimum diameter of the ejector. The four ports were connected in series by a single pipe to a pressure transducer. The transducer output was connected to the recorder interface.

2.4.6.4 RPM. Engine percent RPM was obtained from the engine tachometer output as the period of an electrical frequency. An electronic interface was built to convert tachometer output to a 0 to 5 volt input for the recorder.

2.4.6.5 Indicated Airspeed. Dynamic and static pressure data were obtained from the aircraft pitot static system. A pressure transducer was used to provide airspeed data to the recorder. An airspeed indicator on the photopanel was coupled to the pitot system.

2.4.6.6 Altitude. Pressure altitude data were obtained similarly to the indicated airspeed data. The altimeter system was calibrated from sea level to 40,000 feet for data acquisition purposes.

2.5 BARE ENGINE TEST CELL

2.5.1 Bare engine tests were performed at the Engine Laboratory of the NRC. A J85-CAN-15 engine, serial number 8611 was installed in the No. 5 Ground Level Static Test Cell.

2.5.2 An engine test stand was designed by Orenda Ltd. and fabricated by ComDev. The engine and test stand were bolted to a test bed which in turn was hung from flexible pivots which permit the slight unrestricted movement necessary to activate the load cells. Two load cells, known as a Hagen pneumatic balance load cell and an Emery load cell, were used. Load cell data were displayed in the operator's control room. The engine test stand is shown in Figure 12.

2.5.3 The engine is operated from an adjacent, isolated, control room with a safety window which allows the operator to view the engine. Standard engine instrumentation and engine control systems are located in this room. Special thrust measuring instrument displays were also located in the operation room.

2.5.4 A temporary photopanel was located in the control room and used a 16mm camera to record the operation of the thrust indicator and other engine instruments. A still camera was available to photograph the manometer bank.

2.5.5 A fibreglass bellmouth inlet duct of approximately elliptic contour was installed on the engine. A heavy mesh screen was installed upstream of the inlet bellmouth for safety reasons.

2.6 STATIC AIRCRAFT THRUST STAND

2.6.1 The static thrust stand at the National Aeronautical Establishment of NRC is a T-shaped platform upon which the aircraft is secured as shown in Figure 13. The aircraft nose gear is free to move in the fore and aft direction but is clamped to prevent vertical motion. The aircraft main gear are constrained by the thrust stand platform such that the foreward thrust vector may be measured. This platform is suspended on strap flexures which allow a thrust transfer to two load sensing hydraulic cells. Output from the cells are voltages which register on a remote electrical indicator. The indicator may be selected to display the output of either cell or the electrical summation of both cells.

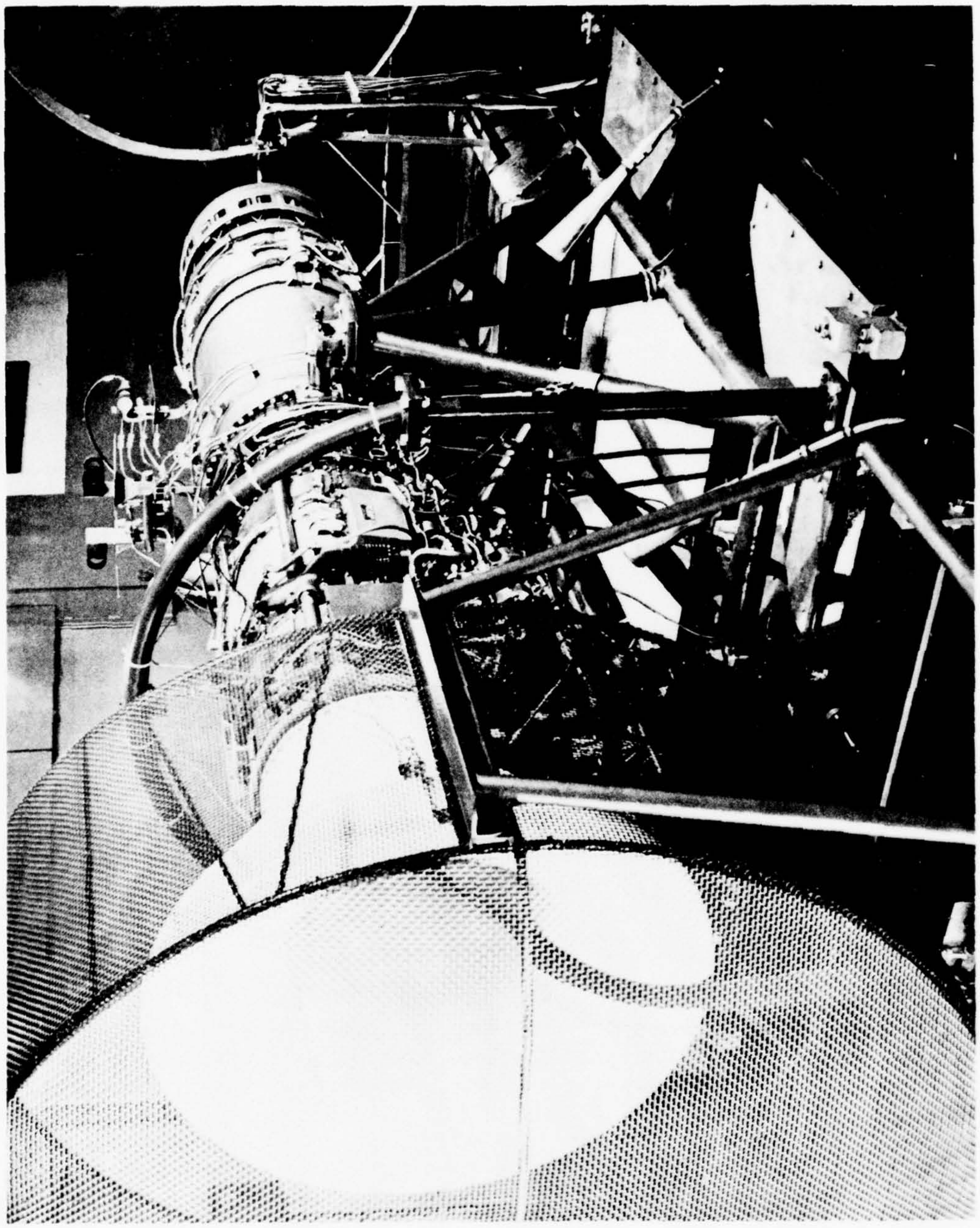


Figure 12: J85-CAN-15 Engine Installed in the NRC Static Test Cell

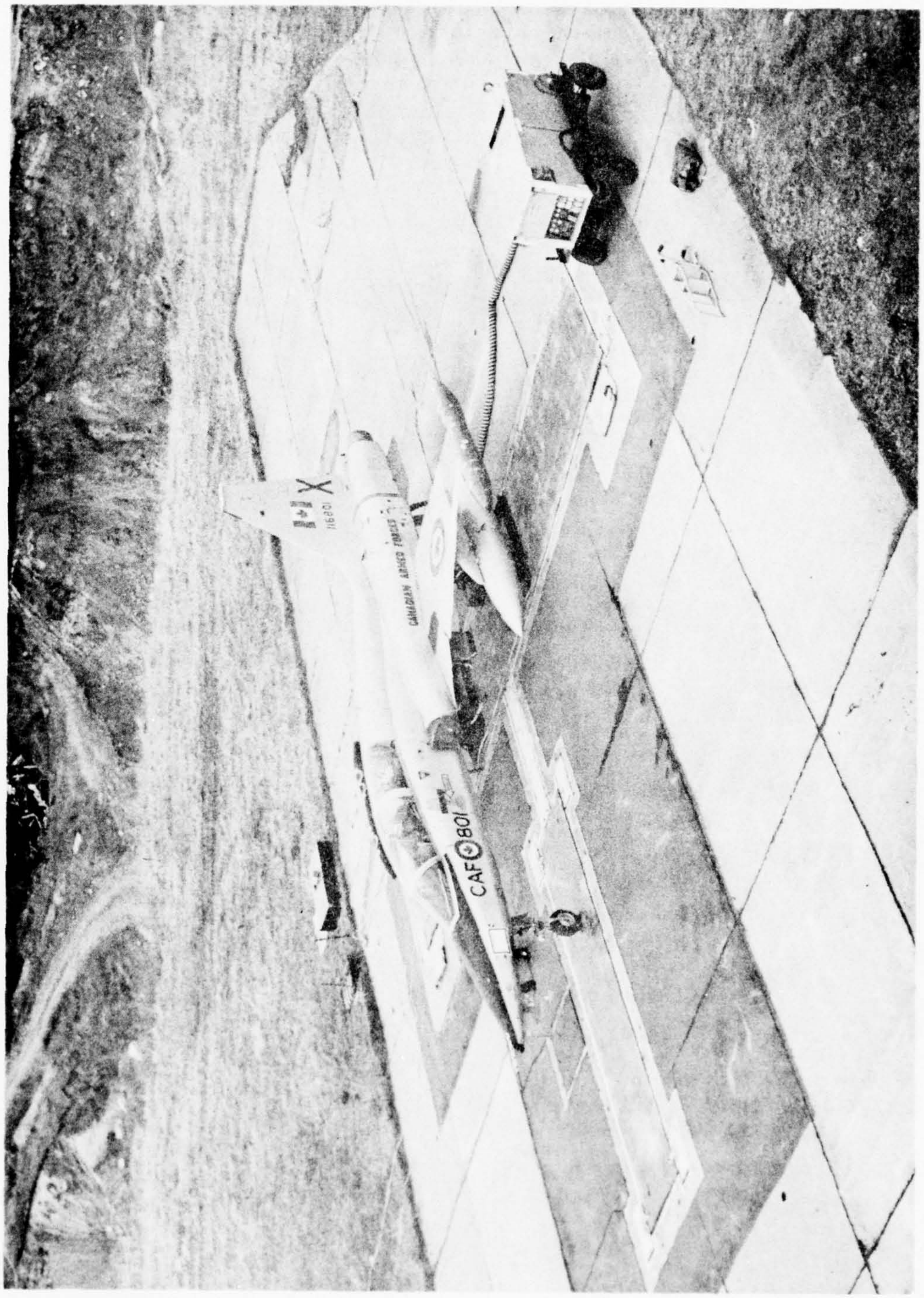


Figure 13: Test Aircraft on the NAE Static Thrust Stand

2.6.2 Continuous data acquisition was accomplished by interfacing the load cells to the Incre-data recorder as shown in Figure 14. Two transducers were used to sense hydraulic pressure on the thrust stand load cells and an amplifier outputs a 0 to 5 volt signal for the data tape recorder. A filter was installed in the electrical circuit to the recorder in order to reduce noise on this circuit. Transducer serial numbers are listed in Appendix XI.

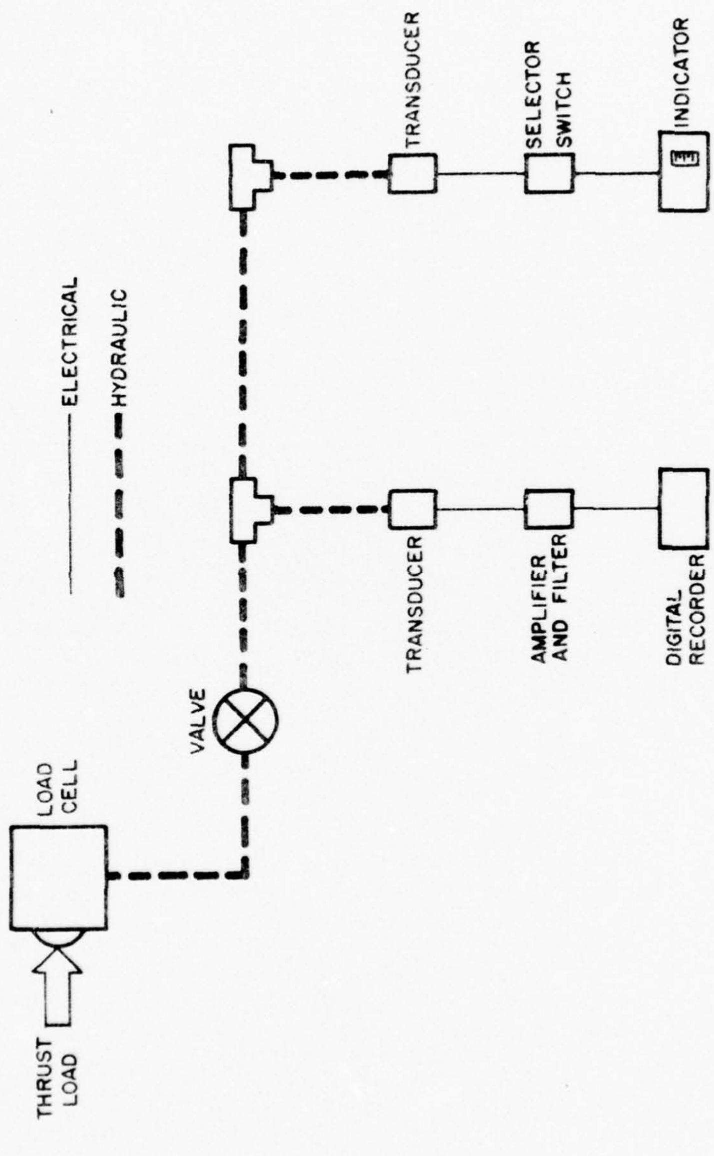


Figure 14: Hydraulic and Electrical Schematic for a Load Cell

SECTION III

TEST PROCEDURE

3.1 BARE ENGINE TESTS

3.1.1 The thrust measuring system was installed on engine serial No. 8611 at the NRC Number 5 Test Cell. CF-5D aircraft type heat shields and insulating blankets were installed to simulate the aircraft thermal environment around the turbine diffuser casing and the upstream part of the afterburner casing. Vibration monitoring devices were installed at the engine bearing locations. Pressure probes and plumbing to the transducers simulated the test aircraft installation. Compensating volumes were added to two pressure lines to achieve desired system transient characteristics.

3.1.2 An Incre-data magnetic tape recorder was located in the operation room and used to record data at one-half second intervals. The following variables were recorded.

(a) Analog output from engine pressure transducers.

(b) Digital output from A/D converter in the TMS computer.

(c) Digital output from the TMS computer for gross and percentage of reference thrust.

(d) Simulated CADC data.

(e) Thrust indicator feedback.

(f) Test stand measured thrust load cell output signal.

3.1.3 A temporary photopanel was located in the control room and used a 16mm camera to record the operation of the thrust indicator and other engine instruments. A still camera was available to photograph the manometer bank.

3.1.4 Prior to a series of engine runs, the NRC engine test cells were calibrated against a secondary proving ring. A hydraulic jack was used to apply a mechanical load through a calibrated proving ring to the test stand load cell. Ring readings were temperature corrected and used to calibrate the indicated load cell readings. Approximately 15 applied loads were used to provide a satisfactory range of data.

3.1.5 The engine was trimmed by adjusting the fuel flow rate to a prescribed value for given compressor inlet conditions at 100% engine RPM as detailed in EO 10B-10E-2, Part 6, Servicing and Maintenance.

3.1.6 Ambient static pressure data were obtained by means of a barometer located within the building but outside of the engine test cell. Test cell pressure depression was obtained from a differential pressure gauge which indicated the pressure drop between the engine cell and the control room which was vented to the atmosphere. Compressor inlet temperatures were read from an instrument in the control room.

3.1.7 Engine runs included steady state power settings in both afterburning and non-afterburning modes. Barometric pressure was recorded before and after each run and the following were recorded during steady state operations: time, test cell pressure depression, compressor inlet air temperature,RPM, P_{S6}, P_{S7}, P_{T5}, throttle position, nozzle position, engine oil pressure, engine mount vibration amplitude, fuel supply temperature, compressor discharge pressure, turbine outlet temperature, engine fuel flow, nozzle exit total pressure and load cell pressure (thrust).

3.1.8 A post-test engine inspection was made to assess the condition of the engine and the TMS hardware in order to ensure the safety of continued testing.

3.2 STATIC AIRCRAFT THRUST STAND TRIALS

3.2.1 Installed static thrust trials were conducted by installing flight qualified tailpipe hardware on engine serial No. 8476. Probes were PFRT tested by NRC subjecting them to a test of 10 hr 42 min duration. The probes and tailpipe were forwarded to AETE for installation on the test engine. (See Reference 4). This engine was installed in AETE's CF-5D aircraft serial No. 116801. The digital data recorder was mounted on a tray in the rear cockpit and used to record engine and flight data at ½ second intervals. Thrust stand load cell data, as well as the data listed in Appendix III were recorded. Conrac pressure transducers were installed for the initial tests and SE transducers used for test NAE-8 and onward. Thrust stand measured thrust data outputs were interfaced with the recorder. A thrust indicator was mounted in the aircraft forward cockpit instrument panel. A second indicator was mounted in the photopanel.

3.2.2 Ground static trials included testing to observe the effect of inlet blockage on thrust. This test required the installation of an inlet area blockage plate as shown in Figure 15.

3.2.3 The NAE static thrust stand load cells were calibrated against a Moorehouse proving ring; serial No. 1331. NAE use a preload cell pressure of approximately 30 psia (1400 lb thrust equivalent) as a zero load state. NAE calibrate the test stand before and after each test run with the aircraft in position. A total of fifteen readings of applied and indicated loads were made with increasing and decreasing loadings. This produces two calibration curves each with a zero load bias reading and a hysteresis loop. It is assumed that any shift in the zero load bias can be corrected as a linear function of time. Therefore, test readings must be correlated to time and the two calibration curves must be interpolated as linear functions of time in order to resolve a single thrust reading. This was accomplished by including the calibration data with the data reduction software program. A zero load bias of about 70 lb/cell ensures platform-cell pressure point contact.

3.2.4 The CF-5D aircraft was operated by qualified AETE personnel. Thrust stand calibrations were performed by NAE personnel and ComDev personnel operated the data recording system.

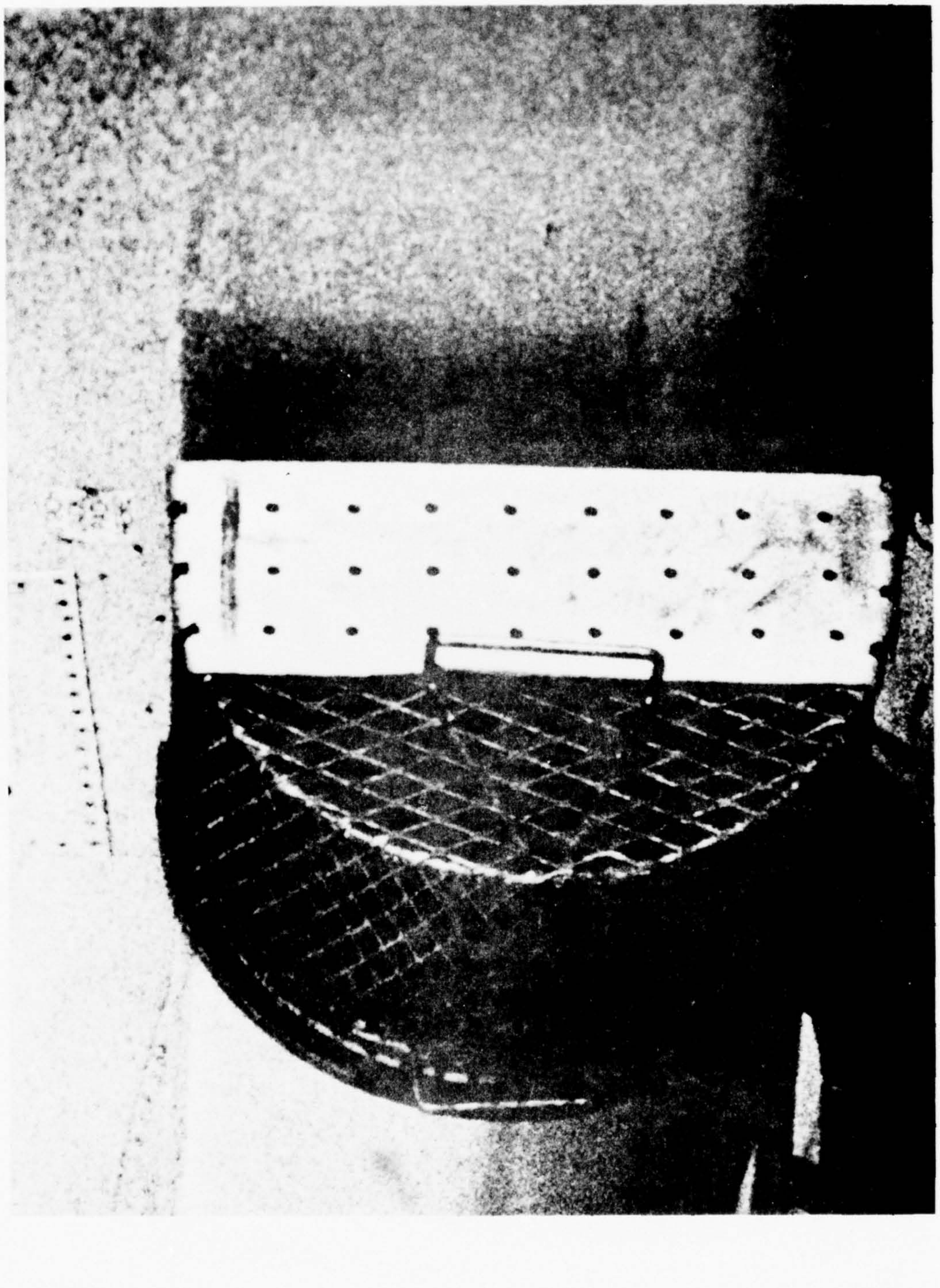


Figure 15: Inlet Area Blockage Plate Mounted on the Test Aircraft

3.2.5 Electrical power was applied to the measuring equipment at least one hour prior to a test. Ambient temperature and barometric pressure as well as wind velocity and direction data were recorded for each test day. These data are listed in Volume II of this report.

3.3 FLIGHT TESTS

3.3.1 Flight trials were conducted with the same engine and aircraft as was used in the static trials. All flights were made with the SE transducers. The TMS computer and data recorder were installed in the rear cockpit. The engine was trimmed at the AETE engine test stand prior to the flight trials. It was retrimmed following an engine removal after FT-03. (The engine was removed to allow an inspection and repair of the P_{S6} probes). A satisfactory aircraft acceptance flight was required prior to commencement of the actual flight trials.

3.3.2 Flight tests were conducted by AETE at Cold Lake, Alberta, in January, February and March 1973. The CF-5D aircraft, serial No. 116801, was used as a test vehicle for the flight trials of the TMS. Engine serial No. 8476 was instrumented for this purpose and used as the starboard engine of the aircraft.

3.3.3 A flight test program was devised by the AETE Project Officer and Project Pilot. Appendix IV of this report is the Final Test Program, Project Directive 71/81, Installation and Demonstration of ComDev Thrustmeter, under which AETE conducted static and flight tests of the TMS. Testing included pre-takeoff engine checks, takeoff performance, climbs and dives, level flight steady state and accelerations, G-loading tests, airstarts, sideslip tests, aircraft configuration changes, landing performance and post landing engine checks. Several flights included Rutowski performance trials. The pilot prepared a knee pad test plan card from the test engineer's detailed flight plan and used this as a guide during the flight. Radio transmitted data were recorded continuously by a voice recorder and in part by the test engineer.

3.3.4 ComDev personnel serviced the digital magnetic tape recorder and performed tests to ensure the satisfactory operation of the TMS prior to each flight. AETE personnel were responsible for aircraft servicing, maintenance and operation.

3.3.5 The data recorder was started as part of the engine start-up procedure. Recording continued until after the post-flight engine tests were conducted. Data processing was usually accomplished within 24 hours of the completion of the flight. An IBM System 360/30 located at Edmonton, Alberta was used for this purpose.

3.4 DATA COLLECTION

3.4.1 During the bare engine running, data were hand recorded by the test personnel who read the instruments and recorded their readings. A camera was used to record manometer levels during certain trials. The films produced were read as a means of cross checking the original data. The digital recorder was used during the TMS testing trials.

3.4.2 Calibration data from the NAE thrust stand trials were manually recorded and transcribed to punched data cards for data processing. A log was used to record ambient conditions and test details. Many thrust stand data readings were recorded in the log for cross checking purposes. The data recorder was used to record engine and thrust stand data throughout most of the trials. The photopanel was used to record sample data.

3.4.3 Flight trial data were collected by the AETE Project Officer, Project Test Pilot (QTP) and the ComDev Project Engineer. A voice recording was made of QTP/Project Officer radio transmissions. A photopanel was used to collect sample data and the digital recorder was used throughout all of the flights.

3.5 DATA PROCESSING

3.5.1 Engine status deck data were acquired by using the General Electric, J85-CAN-15 Computer Deck No. P 15060 (June 1970), with the IBM System 360/67 computer at NRC.

3.5.2 Bare engine data were manually processed with the aid of a programmable calculator. Data processing included the computation of indicated thrust from load cell data and a calibration curve. Gross and reference gross thrusts were computed by the ComDev equation. See Appendices I and II. When the magnetic data recorder was used, data processing was accomplished with the ComDev CDC 6400 computer and software written by ComDev. Automatic data processing included computations of the following:

(a) Gross and reference gross thrust from engine pressure and ambient air data.

(b) Error between (a) above and the TMS output.

(c) Test stand thrust.

(d) The error between (c) above and the TMS output.

(e) The error between the recorder A/D conversions and the TMS A/D converted data.

3.5.3 Static installed engine testing taped data were processed by means of the ComDev CDC 6400 computer and software written by ComDev. An IBM System 360/65 computer was available as a stand-by unit. NAE personnel processed thrust stand calibration data through the IBM System 360/67 at NRC as a cross check on ComDev's data processing program.

3.5.4 Gross and reference thrust and error computations were made as shown in para. 3.5.2 above. Test stand calibration curve computations were included in the data reduction software. Thrust data were manually calculated for check purposes as illustrated by the following example.

(a) The average observed test stand thrust recorded at 11:46 hr., 6 Nov 72, was 2935 pounds. The engine was being operated with the throttle setting at MIL power, auxiliary takeoff doors open, customer bleed off, extracted horsepower non-zero, anti-icing off and T2 heater valve in the open position.

(b) A thrust stand pre-load correction was timewise interpolated from the calibration data as a correction of -209 pounds. Indicated thrust, 2955 lb, plus the pre-load correction, -209 lb, is the indicated net thrust, 2726 lb.

(c) The before and after calibration curves were interpolated to obtain the calibration correction (calibrated net load minus indicated net load) of 65 pounds.

(d) The calibrated propulsive thrust of the test engine was obtained by an algebraic addition of the calibration correction, 65 lb, plus the indicated net thrust, 2726 lb. Therefore the calibrated propulsive thrust was 2791 lb.

(e) Calibrated propulsive thrust is the net thrust as measured by the test stand. It is composed of the engine gross thrust and all engine intake losses as well as ejector losses, etc.

3.5.5 Flight test data were processed at Edmonton, Alberta, by means of an IBM System 360/30 computer and software modified for this computer. Aircraft variant measurements were added to the software program and thrust stand data reduction procedures were deleted. A copy of the flight trial data reduction software was maintained in an up-to-date status for the ComDev CDC 6400 computer as a back-up to the Edmonton computer.

3.5.6 Data tapes were processed in Ottawa using the IBM System 360/67 computer and plotter at NRC in order to produce plotted test results. Plotted data are included in Volume II .

SECTION IV

SUMMARY OF TESTS

4.1 CHRONOLOGY AND STATISTICS

4.1.1 A series of bare engine tests was started in December 1969. Engine runs were designated only by run numbers. They were conducted mainly to obtain calibration data and to test the suitability of engine hardware for flight use. Table I presents a chronology of the J85-CAN-15 serial No. 8611 engine tests in the NRC test cell.

Table I Chronology of J85-CAN-15 No. 8611 Engine Cell Tests

Engine Run Number	Purpose Of Test	Accumulated Run Time (Hr:Min)	Instrument Set	Start Date	End Date
0-28	Engine test stand setup	(no data)	separate probes	Dec 69	Feb 70
29-39 40-46	Calibration data collect-ion	9:57 5:29	separate probes	Feb 70	Apr 70
47-57	Engine Hardware development	30:35	separate probes	May 70	Jun 70
58-61	Engine Hardware development	31:21	separate probes	Mar 71	Mar 71
62-84	Engine hardware development	49:17	manifold probes	Jun 71	Aug 71
85-108	PFRT of first flight probe instrument set	73:16	1^{st} Flight probes designated 1-1-1	Oct 71	Nov 71
109-160	Development & testing second P_{S6} probes and P_{S7} probes & first P_{S7b} probe	99:21	Second probes Designated 1-2-2-1	Dec 71	Mar 72
161-181	Acceptance test on final instru-ment set	110:03	final set designated 1-2-0-2	Mar 72	May 72

4.1.2 Installed static tests at NAE were started 1 November 1972. Test runs were designated by number and prefixed with the letter code NAE. Runs NAE1 to NAE8 were made prior to the flight trials and runs NAE9 to NAE12 were made as post-flight test calibrations. Table II presents a chronology of these trials.

Table II Chronology of J85-CAN-15 No. 8476 Engine Static Tests Installed in CF-5D Aircraft Serial No. 116801

RUN	DATE	PURPOSE OF TEST	RUN TIME (Hr:Min)
NAE-1A	1 Nov 72	Test system shakedown	00:45
NAE-1B	1 Nov 72	Test system shakedown Nose wheel clamp installed	00:40
NAE-1C	1 Nov 72	Nose wheel clamp & oleos deflated	00:20
NAE-2	6 Nov 72	Steady state runs, auxiliary takeoff doors open and closed	01:23
NAE-3	6 Nov 72	As NAE-2. First data run with tape recorder	01:25
NAE-4	10 Nov 72	As NAE-3.	01:18
NAE-5A	11 Nov 72	Intake area blockage plate installed	01:05
NAE-5B	11 Nov 72	Intake area blockage plate removed	00:40
NAE-6	13 Nov 72	Power transient trials	00:40
NAE-7	17 Nov 72	Steady and power transient trials	00:46
NAE-8	20 Nov 72	As NAE 2,3,4 and 7 except SE transducers used.	01:26
—	—	(break for Flight Trials)	—
NAE-9	28 Mar 73	Steady state runs	01:40
NAE-10	30 Mar 73	Steady and power transient trials	01:13
NAE-11A	5 Apr 73	Intake area blockage plate installed	01:00
NAE-11B	5 Apr 73	Intake area blockage plate removed	00:58
NAE-12	6 Apr 73	Dynamic and steady state trials	01:12

4.1.3 Flight trials were started at AETE, Cold Lake, Alberta, with an acceptance flight on 18 Jan 1973. Flights were designated by number. Acceptance flights were prefixed AE and flight tests were prefixed FT. Table III presents a chronology of the flight trials.

Table III Chronology of J85-CAN-15 No. 8476 Engine Flight Tests
Using CF-5D Aircraft Serial No. 116801
(Sheet 1 of 2)

FLIGHT	DATE	PURPOSE OF FLIGHT	FLIGHT TIME (Hr:Min)
AE-1	12 Oct 72	Aircraft acceptance flight.	00:47
AE-2	25 Oct 72	Acceptance flt prior to ferry to Ottawa.	00:45
AE-3A	16 Jan 73	Acceptance flight following ferry to Cold Lake. Aircraft unserviceable for flight.	00:00
AE-3B	18 Jan 73	Acceptance flight following aircraft repair. CADC reported unserviceable.	00:35
AE-4	26 Jan 73	Acceptance flight following CADC repair.	01:00
FT-01	29 Jan 73	Flight test. Max power takeoff and level flight performance test.	00:42
FT-02	30 Jan 73	Flight test. MIL power takeoff and level flight performance test.	00:57
FT-03	31 Jan 73	Flight test. Engine transient and level flight performance test.	00:41
FT-04	8 Feb 73	Flight test. Rutowski minimum fuel climb to 45K ft. Engine performance at 45K ft. Airstarts at 20K ft.	00:46
FT-05	9 Feb 73	Flight Test. Modified Rutowski minimum fuel climb to 30K ft. Level flight trials at 30K ft.	00:41
FT-06	12 Feb 73	Flight test. Modified AOI climb to 32K ft. Constant Mach number performance trials.	00:52
FT-07	13 Feb 73	Flight test. Climb with 94% RPM on port engine and MIL power on starboard engine. Engine thrust comparison and constant Mach number performance trial.	00:53

Table III Chronology of J85-CAN-15 No. 8476 Engine Flight Tests
Using CF-5D Aircraft Serial No. 116801
(Sheet 2 of 2)

FLIGHT	DATE	PURPOSE OF FLIGHT	FLIGHT TIME (Hr:Min)
FT-08	14 Feb 73	Flight test. Max power climb and maximum airspeed (V_{MAX}) dive. Effect of G-loading on TMS.	00:41
FT-09	15 Feb 73	Flight test. Rutowski minimum time to climb profile. Effect of sideslip on TMS performance. Drag evaluation using the TMS.	01:00
FT-10	15 Feb 73	Flight test. Maximum power climb, AOI schedule climb. Level flight performance.	00:39
FT-11	8 Mar 73	Flight test. Minimum fuel climb profile to 35K ft. Afterburner light at 45K ft. Airstarts at 20K ft.	00:40

SECTION V

RESULTS

5.1 GROUND TESTS

5.1.1 Bare Engine Tests

5.1.1.1 P_{T5} total pressure probes, as shown in Figure 5, were designed and a PFRT was performed to demonstrate design safety. P_{T5} probes were designed such that they could be easily installed or removed from the engine diffuser struts.

5.1.1.2 P_{S6} static pressure probes as shown in Figure 3 , were designed and PFRT proven. Probes had a low profile to the engine gas flow. A flexible expansion tube was incorporated to account for thermal expansion and afterburner liner displacement relative to the casing.

5.1.1.3 P_{S7} static pressure probes, as shown in Figure 4, were designed and PFRT proven in the b location. The "b" location is the second last row of liner bolt holes and the "a" location is the last row of bolt holes. These probes and their plumbing are satisfactorily clear of the VEN and use existing liner bolt holes.

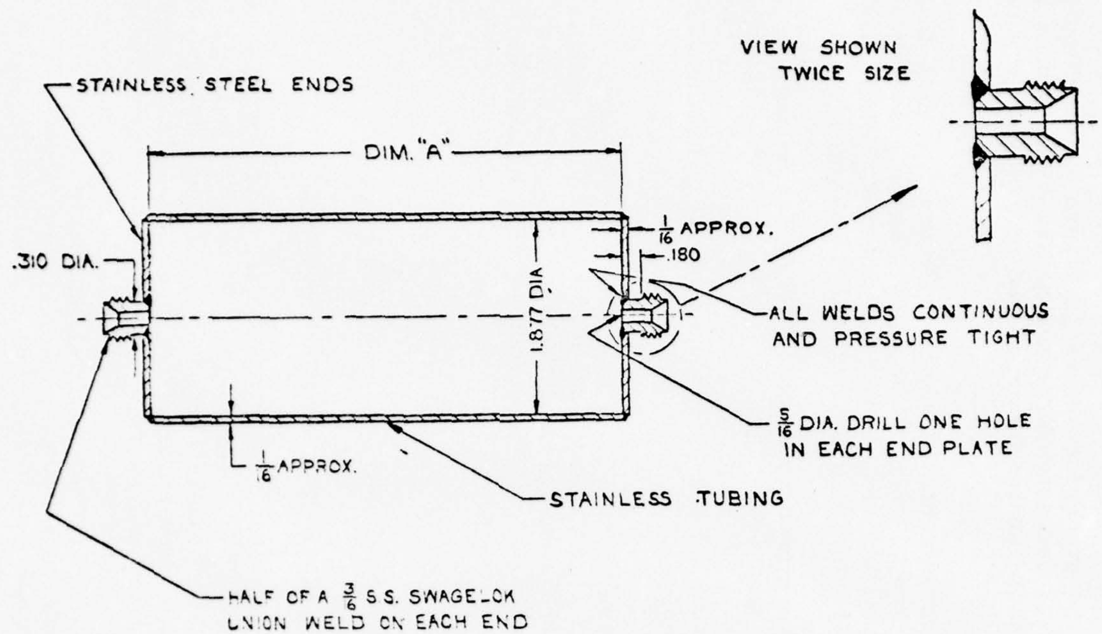
5.1.1.4 Engine running tests provided a means of testing various probe designs and pointed out design faults. Reference 4 provides further test details. Probes, of the design used in the flight tests, were serviceable after the completion of the following PFRT engine running times.

(a) P_{T5}, total pressure probe, 60 hr 46 minutes.

(b) P_{S6}, static pressure probe, 36 hr 47 minutes.

(c) P_{S7}, static pressure probe, 10 hr 42 minutes.

5.1.1.5 Bare engine running indicated the need for delay volumes in two of the engine pneumatic pressure lines to the TMS transducers. It was observed that during rapid increases or decreases of power between idle and MIL, the computed gross thrust consistently fluctuated to a high value for a ½ second computer update.This effect did not occur during steady state running. The electrical output of the three engine pressure transducers were monitored and it was determined that pressure signals were arriving at the transducers out of phase. Delay volumes, as shown in Figure 16, of 12 cubic inches in the P_{T5} line and 6 cubic inches in the P_{S6} line, eliminated large pressure differentials and resulted in acceptable TMS performance.



FOR	No. REQ'D	DIM. "A"
PTE	ONE ONLY	4.337
PS6	ONE ONLY	2.168

Figure 16: Delay Volume Design

5.1.1.6 Calibration constants C_{5-6} C_{6-7} and E which account for deviations between the one dimensional analysis theory and the actual engine, were obtained using static bare engine test data. A total of 17 afterburner and 72 non-afterburner bare engine cell indicated thrust data points from runs 29 to 39 (9 hr 57 min engine running) were used in computing the calibration constants.

5.1.1.7 Bare engine test data from Runs 29 to 46 were used in demonstrating the repeatability capability of the TMS. Measured thrust, as indicated by the NRC engine test cell, is compared with gross thrust TMS equation solutions. A full scale (F.S.) thrust has been defined as 4300 pounds. This value is the bare engine static rated maximum afterburner power at sea level standard day conditions. This definition provided a deviation band about the line of perfect agreement shown on many figures. Figure 17 shows 122 data points (all available data) obtained from the engine test cell indicated thrust and the TMS computed thrust for runs 29 through 46. The data include a power range of 368 to 4171 lbs. The standard deviation of this data set is ±34.6 lbs. Figure 18 shows similar data taken from the magnetic tape recording of Run 149. These data were selected at random to include a thrust range of 300 to 4700 lbs.

5.1.1.8 Bare engine test data from Run 149 were used to demonstrate the tracking capability of the TMS. The engine power lever angle was advanced such that the engine RPM was held at selected settings for 10 second durations. RPM was increased in this manner from 70% to 100%. Figure 19 shows the resulting data from this test. A second tracking test consisted of a rapid advance (slam) of the power lever from idle to maximum afterburner followed by reductions in power to intermediate and minimum afterburning. Figure 20 presents the recorded data for Run 149 for this tracking test.

5.1.1.9 A set of probe pressure leak test gear was fabricated. This facilitated the leak testing of all pressure probes without removing the engine from the aircraft or the probes from the engine.

5.1.2 Static Thrust Stand Tests: (Pre-Flight Tests)

5.1.2.1 Static thrust stand runs NAE 1 to NAE 8 resulted in 10.4 hours of engine running. Six runs yielded digital data and all yielded photo-panel films. A set of data tables is included in Volume II .

5.1.2.2 A description of the method of computing static thrust stand calibrated thrust is presented in Section III of this volume. An additional correction must be made to account for the engine ejector force in order to make a direct comparison between the test stand indicated thrust and the TMS indicated gross thrust. A detailed analysis of the method of accounting for ejector force is shown in Appendix V.

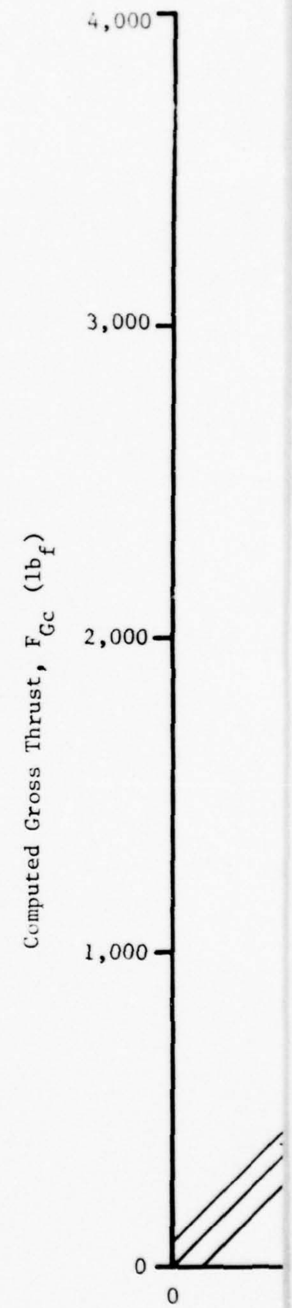
5.1.2.3 Intake blockage was induced by partially blocking the side fuselage inlet duct by means of the intake area blockage plate. This resulted in a decrease of thrust measured by the thrust stand. The decrease for the MIL power case with auxiliary takeoff doors open was 2% of the point. This decrease was typica-ly 6% of the point with the doors closed. The TMS indicated a similar thrust decrease.

5.1.2.4 Actuating the auxiliary takeoff doors from closed to open resulted in a thrust stand static gross thrust increase of approximately 15% at MIL power with the inlet area blockage plate off. The TMS indicated this increase.

5.1.2.5 Repeatability and tracking capabilities of the TMS were demonstrated by plotting data points from runs NAE 2 to NAE 8 in Figure 21. Thrust stand measurements were corrected for the ejector force as explained in Appendix V. A total of 203 data points (all available hand recorded data) have been included from the various runs, over a thrust range of 150 to 4160 pounds, afterburner on and off, auxiliary takeoff doors open and closed and with the inlet area blockage plate on and off. The standard deviation of this data set is $\pm 52.8\ lb_f$.

5.1.2.6 Run NAE 8 was noteworthy because the Conrac pressure transducers were replaced by transducers manufactured by S.E. Laboratories, Feltham, UK. TMS performance was statistically improved for run NAE 8 since the standard deviation of these data is $\pm 23.2\ lb_f$ based upon 33 data points. The SE transducers were used in the subsequent flight tests and post-flight static thrust stand trials.

5.1.2.7 During run NAE 6, one of the P_{T5} probes developed a plumbing leak within the diffuser strut. The leak was discovered during the routine post-run leak testing made on all the engine pressure probe systems. No deterioration in TMS performance was observed. The probe was not repaired until the aircraft was returned to AETE, Cold Lake.


4,000
3,000
2,000
1,000
0
0
Computed Gross Thrust, F_{Gc} (lb_f)

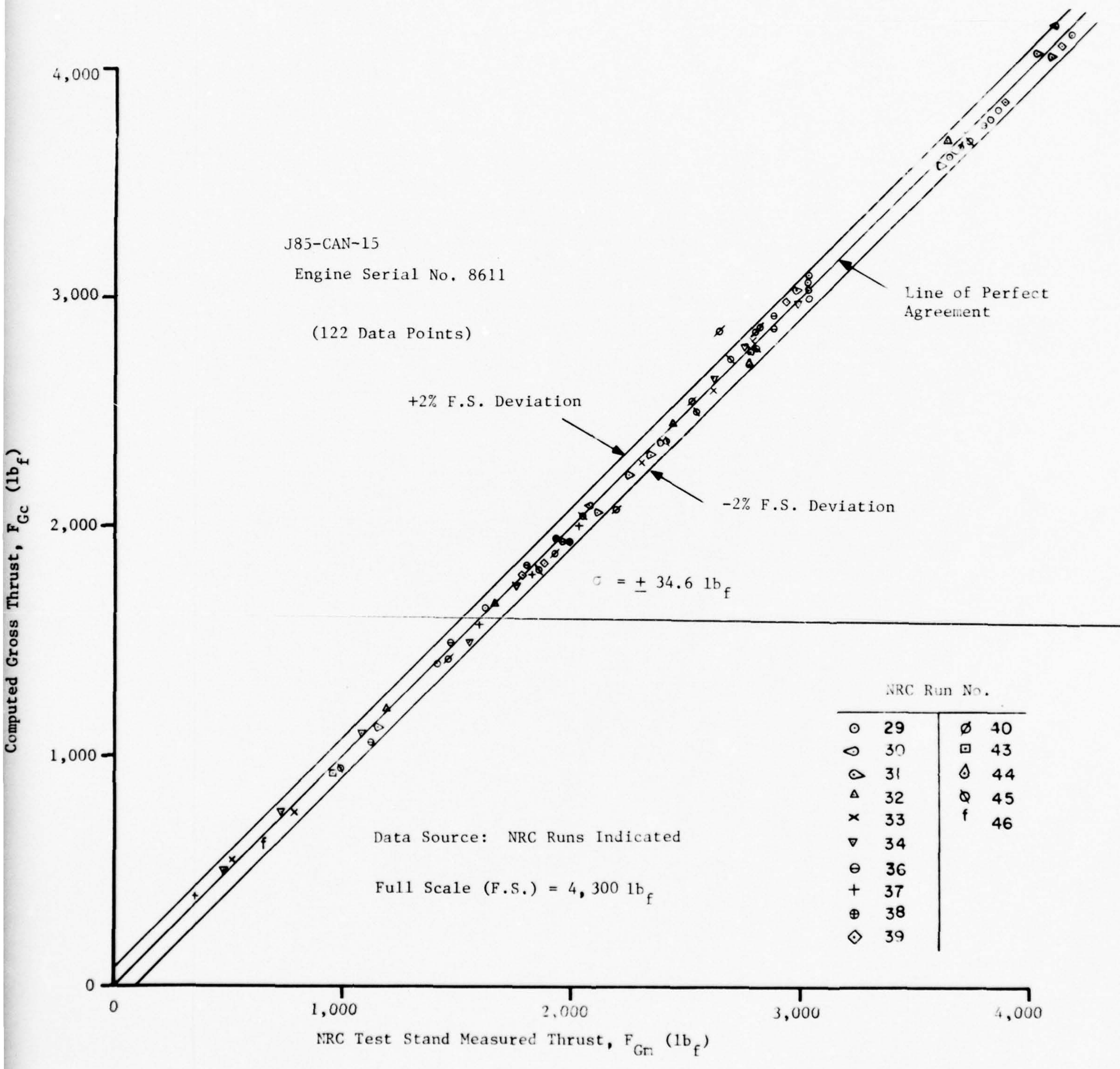


Figure 17: Computed Gross Thrust vs Test Stand Measured Thrust

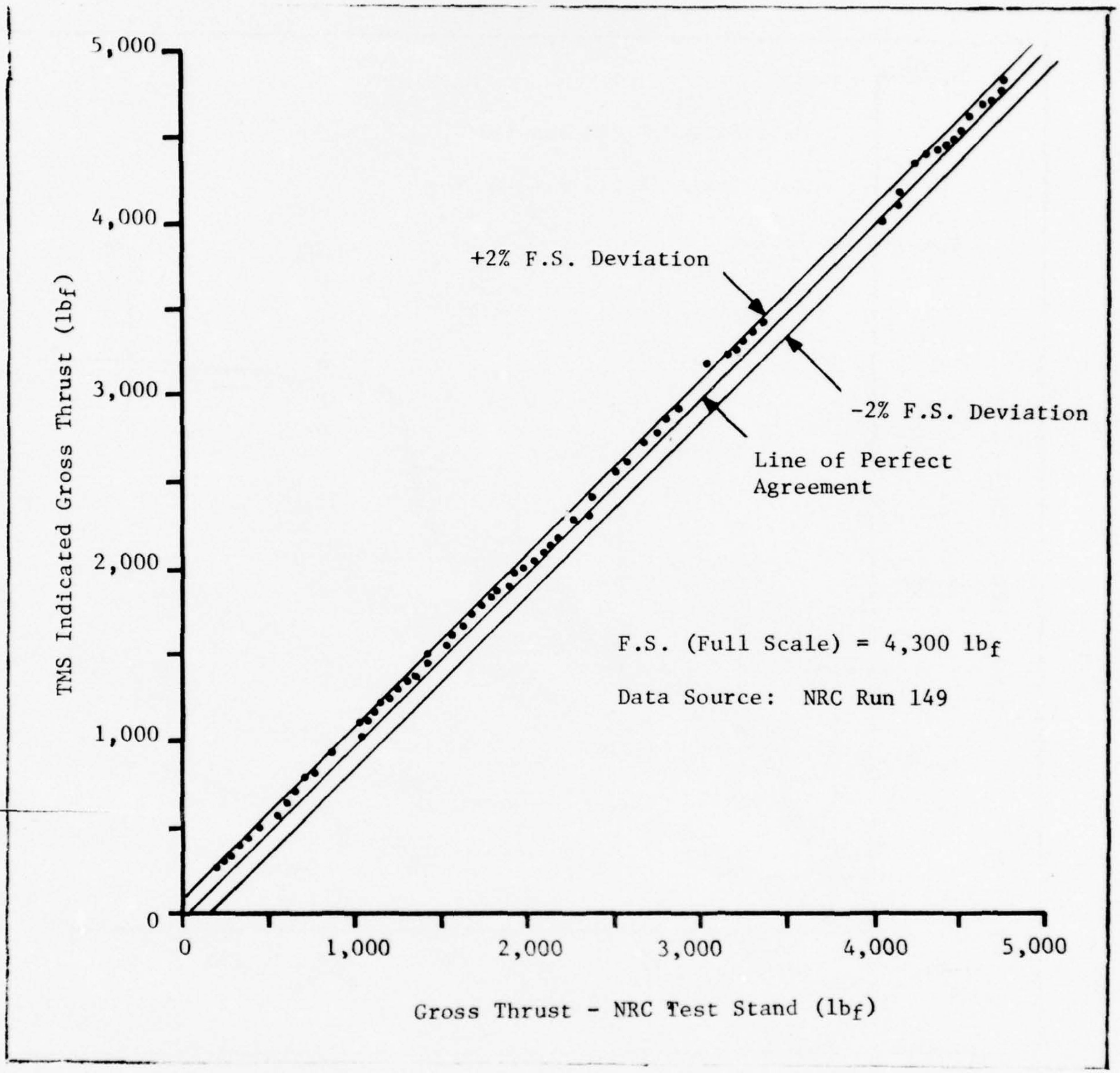


Figure 18: TMS Computed Thrust vs NRC Test Stand Measured Thrust

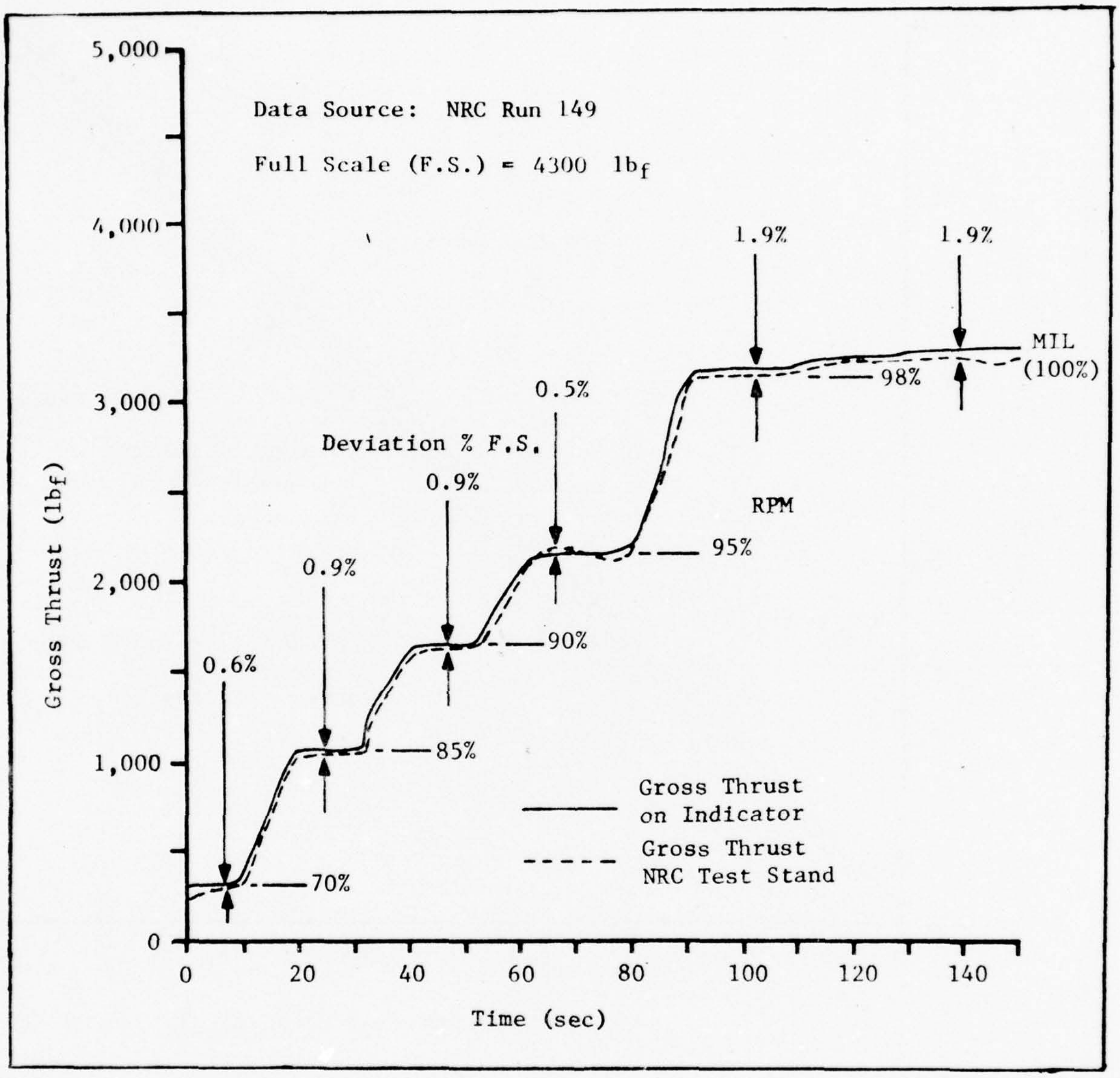


Figure 19: Tracking Capability of TMS - Incremental RPM Advance

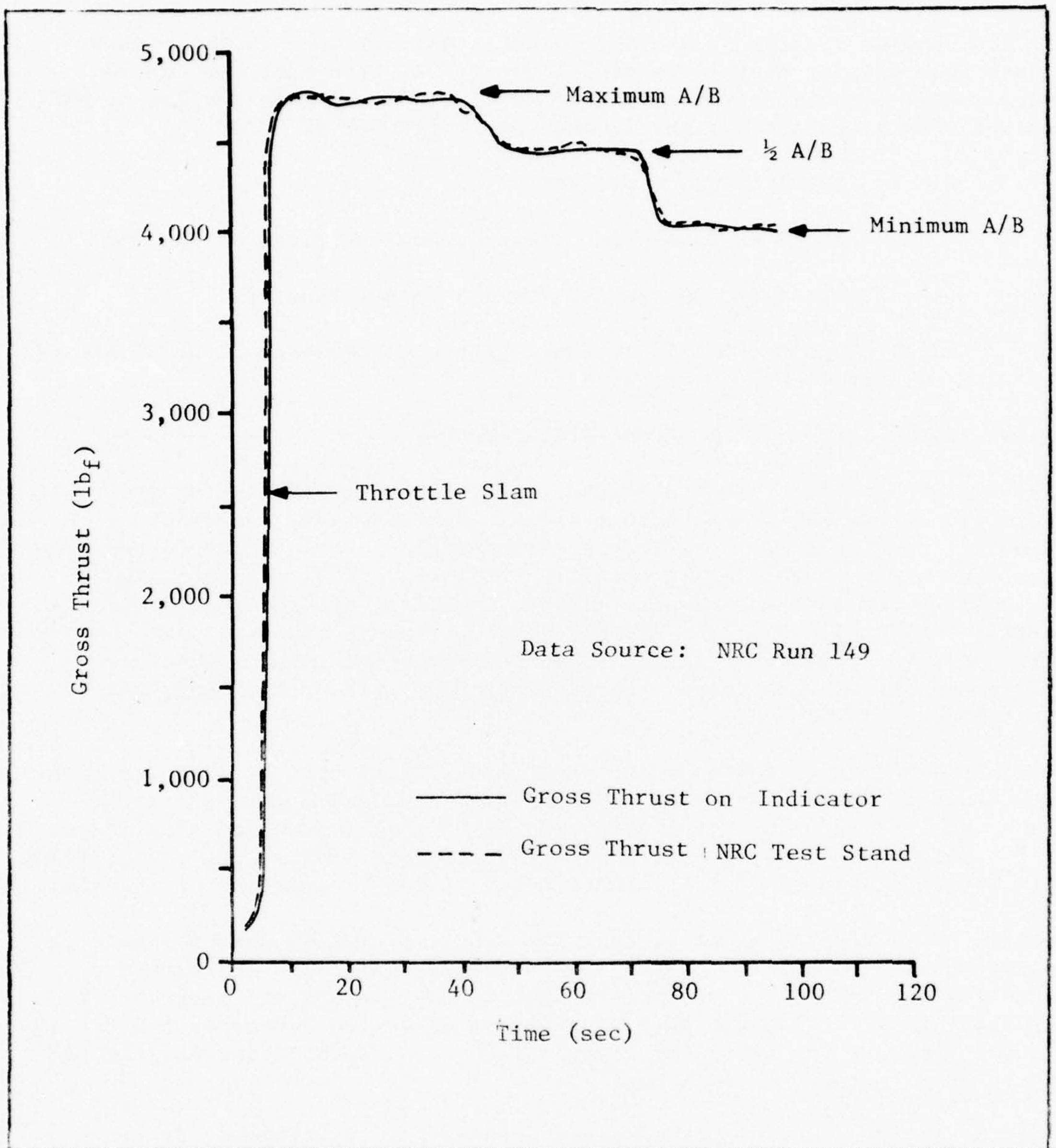


Figure 20: Tracking Capability of TMS - Throttle Slam Trial

5.1.2.8 Engine running at the NAE thrust stand resulted in the pressure probes accumulating test lives as follows. Test life excludes engine running time for trim purposes and 10 hr 42 min acceptance running by NRC, but includes acceptance flight times. See Reference 4.

(a) P_{T5} total pressure probes:

(i) 9 hr 08 min for the probe which failed on NAE 6

(ii) 12 hr 00 min for three probes without failure.

(b) P_{S6} and P_{S7} static pressure probes accumulated 12 hr 00 min of service without failure.

5.1.3 Static Thrust Tests :(Post-Flight Tests)

5.1.3.1 A post-flight series of installed static thrust testing was performed at the NAE static thrust stand. Repeatability and tracking capabilities of the TMS were further demonstrated. All hand recorded data from runs NAE 9 to NAE 12 are shown in Figure 22. This figure includes 145 points over a thrust range of 120 to 3900 lb_f, afterburner on and off, auxiliary takeoff doors open and closed and with the blockage plate on and off. Only 45 points were actually plotted since the other 100 points overlapped the plotted ones. The standard deviation of this data set is ±25.4 lb_f.

5.1.3.2 TMS dynamic responses to power lever transients were indicated by data obtained during run NAE 10. Figures 23 and 24 present NAE thrust stand and TMS computed thrust data recorded during two engine slam tests. As PLA data were obtained only at 3 second time intervals, photopanel films were refered to in estimating a more exact time of initiation of the slam.

5.1.3.3 Data were obtained to show the effect of intake area blockage on engine thrusts both with the auxiliary takeoff doors open and closed. Data from run NAE 11 are shown in Figure 25 for the TMS and in Figure 26 for the NAE thrust stand. NAE thrust stand data were not corrected for the ejector losses. The effect of intake blockage on cockpit instrumentation may be seen in the data of Table IV.

5.1.3.4 All pressure probes remained serviceable through this final series of tests. Probes accumulated life times as follows. Life excludes engine running for trim purposes, ferry flights (approximately 3.5 hr per flight) and PFRT testing (10 hr 42 min applicable only to the P_{S7} probes).

(a) P_{T5} total pressure probes replaced after NAE 8; 16 hr 10 min.

(b) P_{S6} static probes replaced after FT-10; 6 hr 43 min.

(c) P_{S7} static pressure probes; 28 hr 10 min.

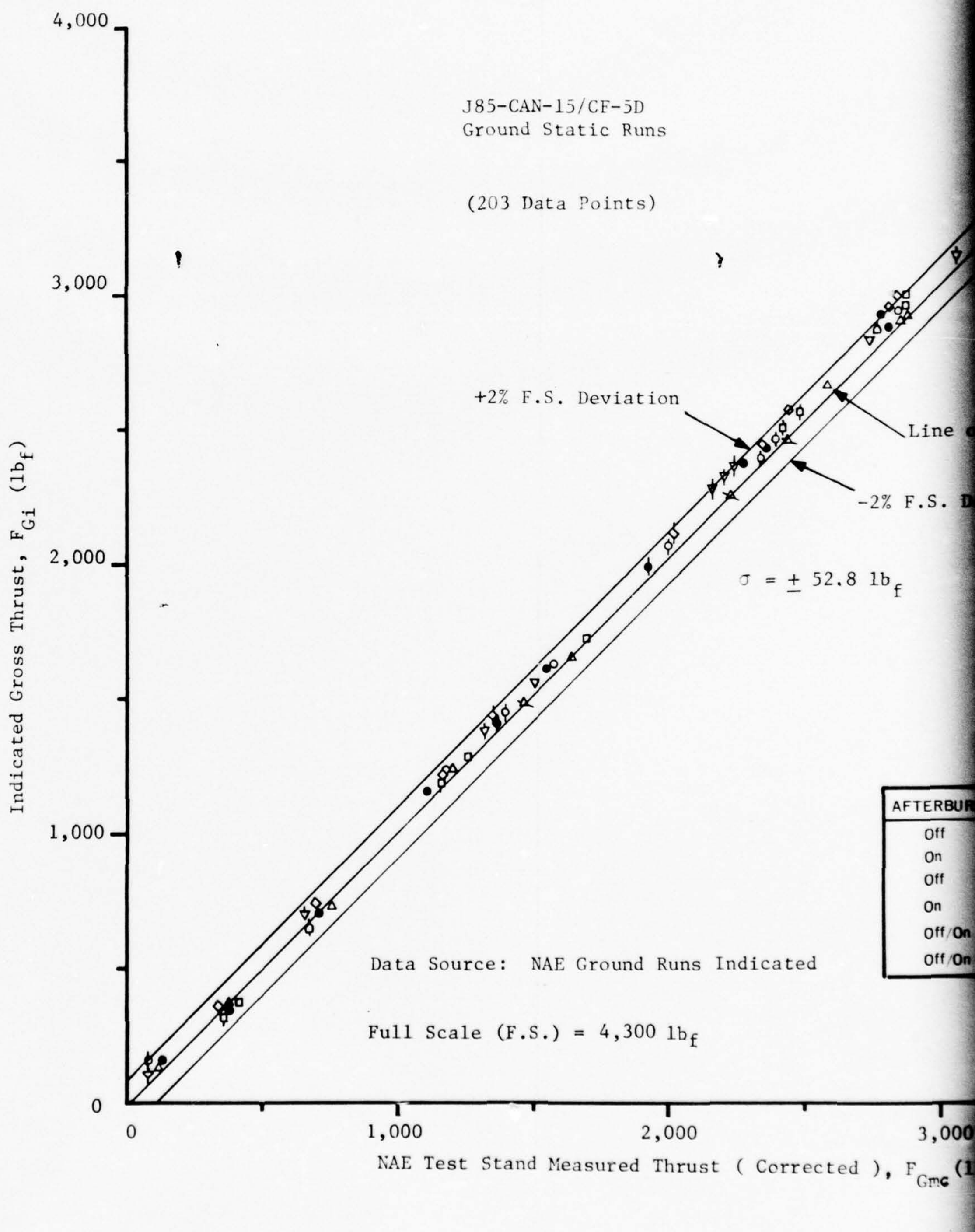


Figure 21: Indicated

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Section V

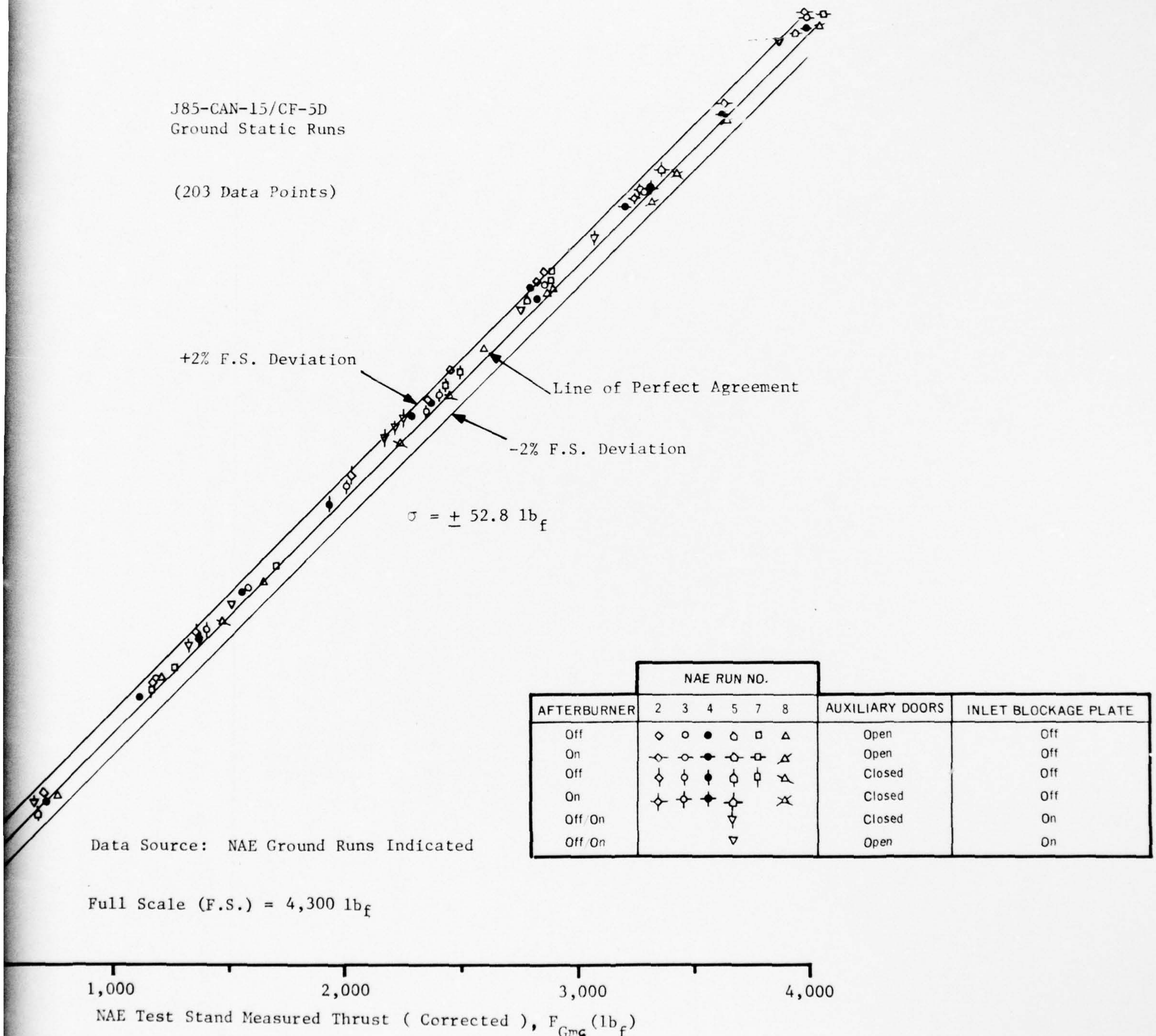


Figure 21: Indicated Gross Thrust vs Test Stand Measured Thrust(Corrected)

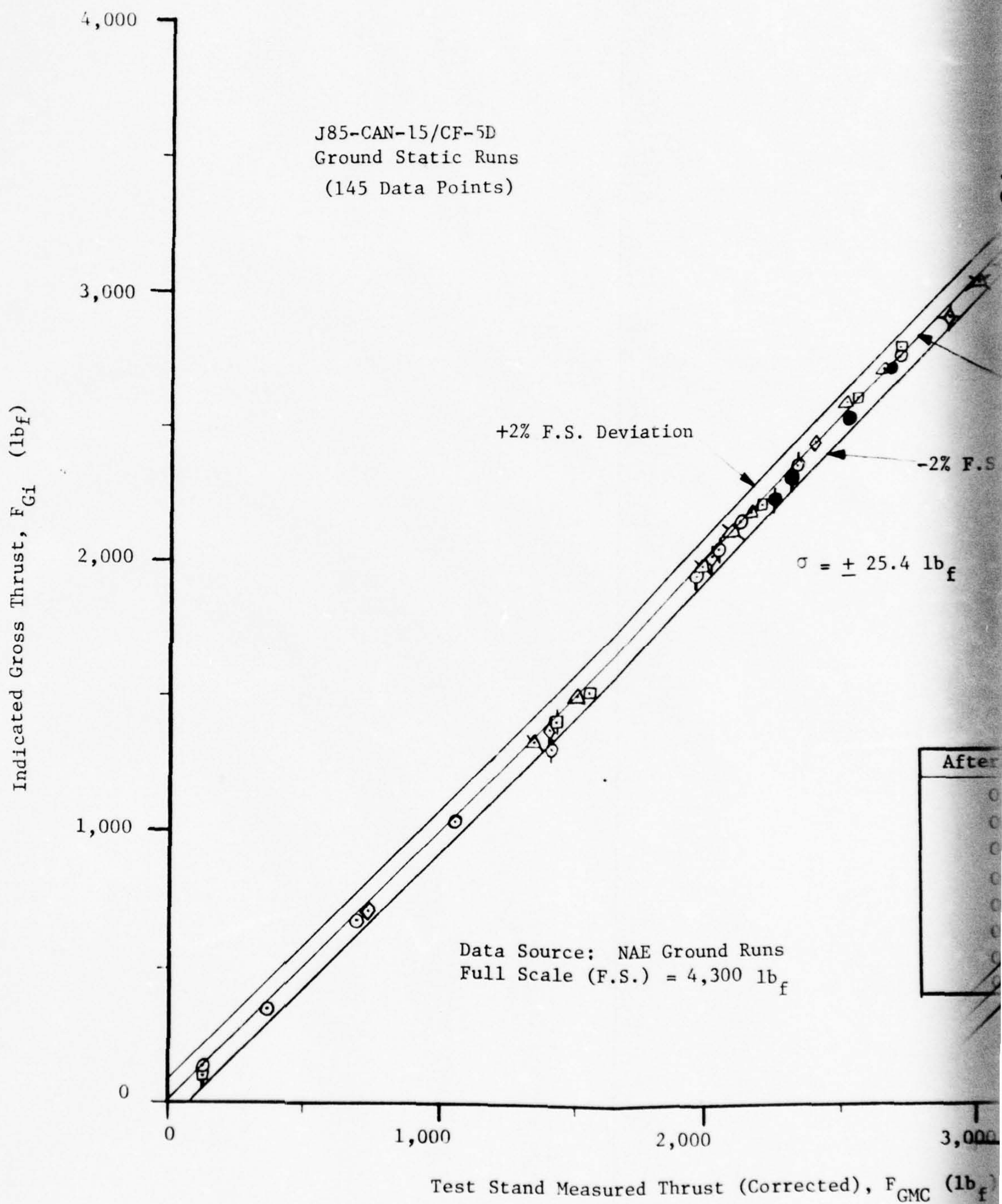
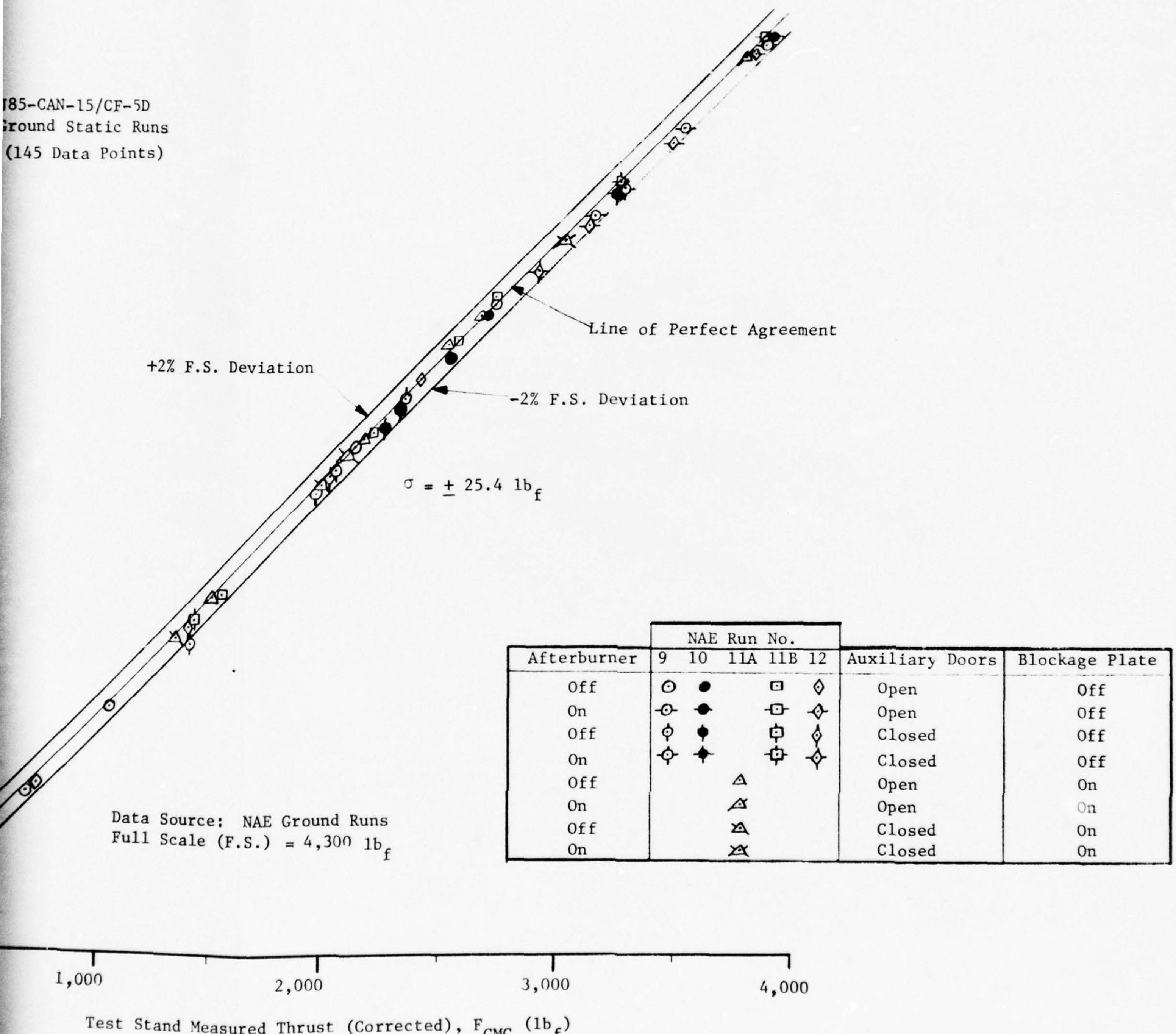


Figure 22: Indic



Afterburner	NAE Run No. 9	10	11A	11B	12	Auxiliary Doors	Blockage Plate
Off	⊙	●		⊡	◇	Open	Off
On	-⊙-	-●-		-⊡-	-◇-	Open	Off
Off	⊙	●		⊡	◇	Closed	Off
On	⊙	●		⊡	◇	Closed	Off
Off			△			Open	On
On			△			Open	On
Off			△			Closed	On
On			△			Closed	On

Figure 22: Indicated Gross Thrust vs Test Stand Measured Thrust (Corrected)

Table IV Effect of Intake Blockage on Cockpit Instrument Indications

CONDITIONS				Effect on Cockpit Instrument Indications					
Aux.Takeoff Doors	Anti-Ice	Cabin Pressure	Intake Blockage	% Ref Thrust	△% Ref Thrust	△Fuel Flow-lb/hr	△Fuel Flow %	NPI % Open	△NPI % Open
Open	Off	Off	Off	98.7	0	0	0	18.9	0
Open	Off	Off	On	95.8	2.9	50	1.6	18.5	0.4
Open	Off	On	Off	94.2	4.5	9	0.3	20.	1.1
Open	On	Off	Off	90.6	8.1	113	3.6	20.3	1.4
Open	On	On	Off	89.0	9.7	110	3.5	20.3	1.4
Closed	Off	Off	Off	82.5	16.2	337	10.7	20.3	1.4
Closed	Off	On	Off	79.2	19.5	321	10.2	22.	3.1
Closed	On	Off	Off	77.4	21.3	372	11.8	22.	3.1
Closed	Off	Off	On	76.3	22.4	422	13.4	22.	3.1

Notes:
1. RPM remained at 100% in all cases.
2. Table IV is based upon all available data, 103 points from NAE testing.
3. △ values are the change with respect to the first auxiliary takeoff doors open case.

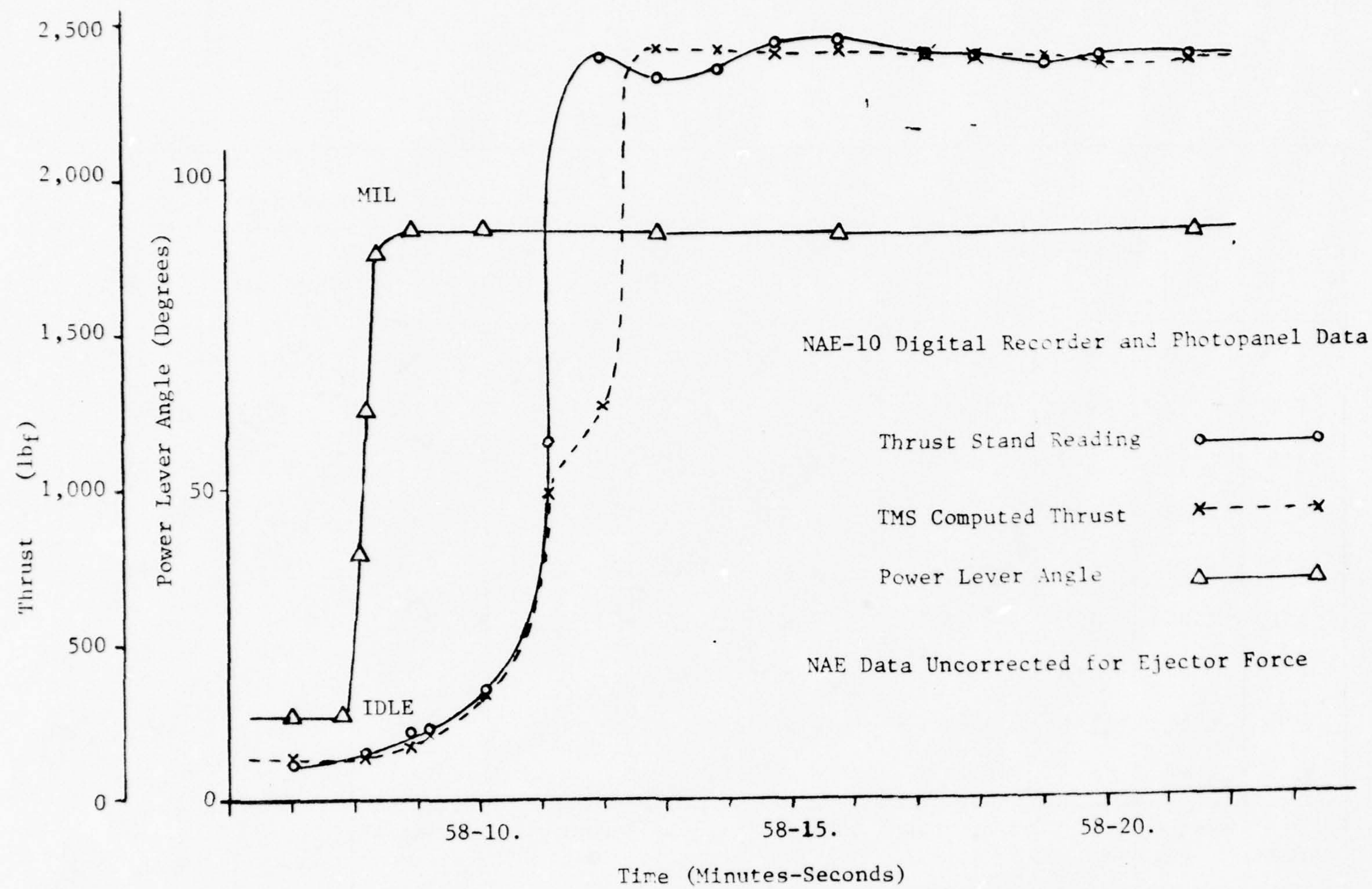


Figure 23: Dynamic Response of TMS On Installed Engine On NAE Thrust Stand

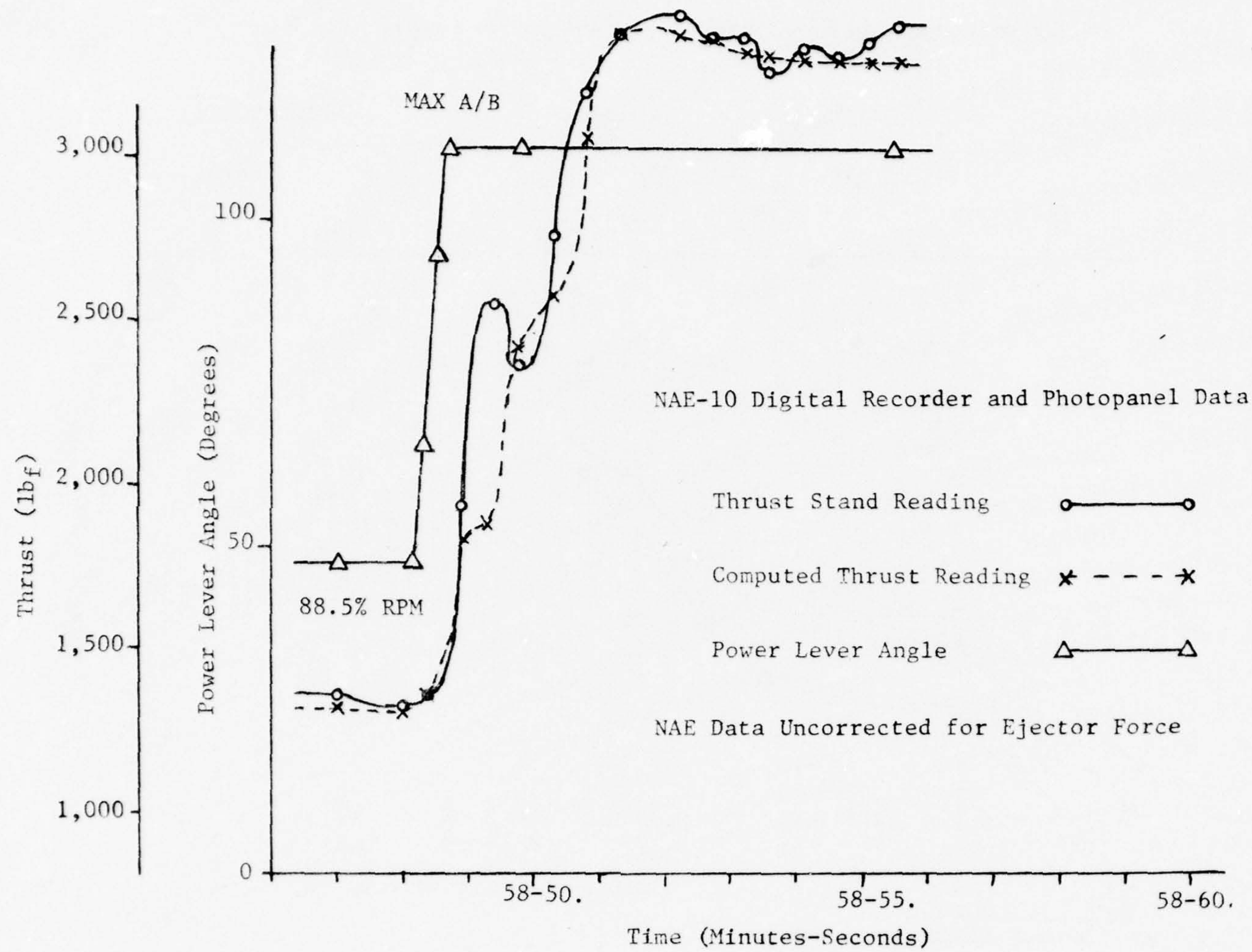


Figure 24 Dynamic Response of TMS On Installed Engine On NAE Thrust Stand

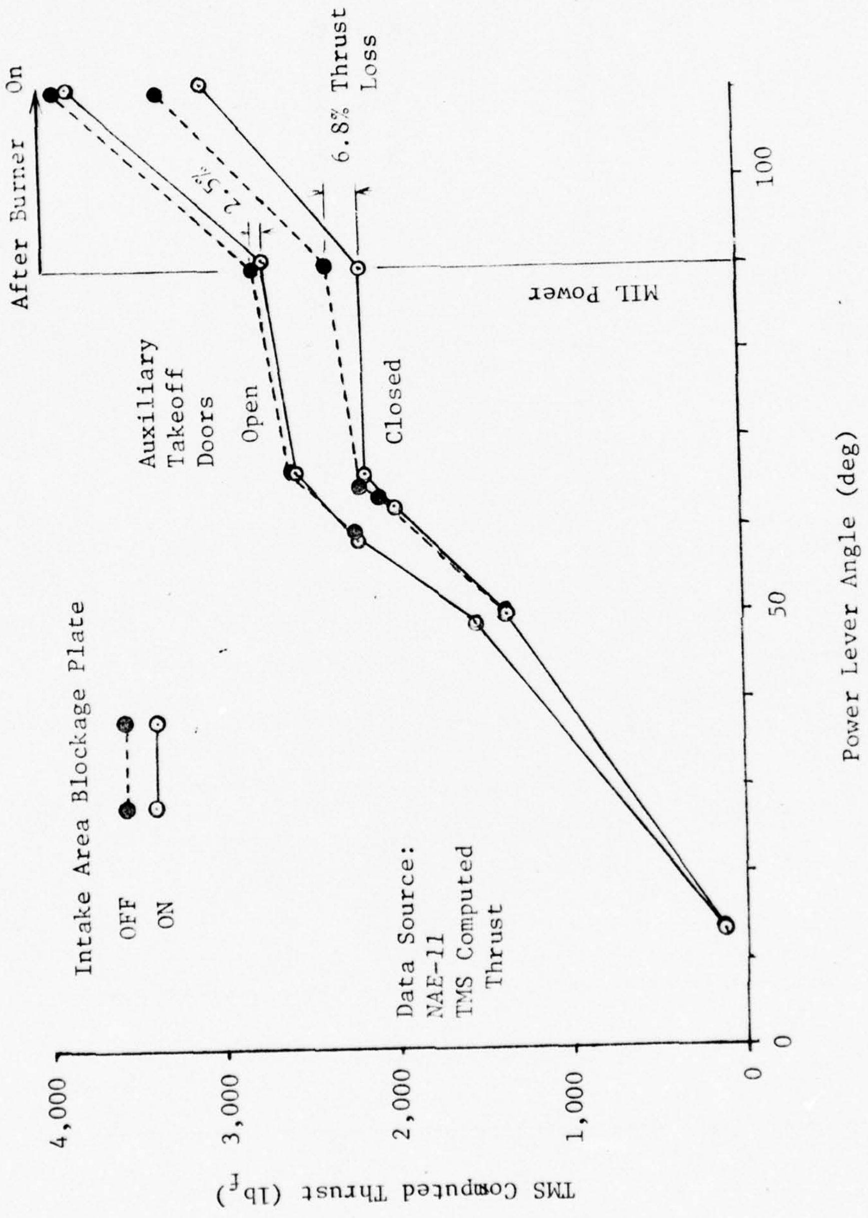


Figure 25: Effect of Intake Area Blockage on TMS Computed Thrust

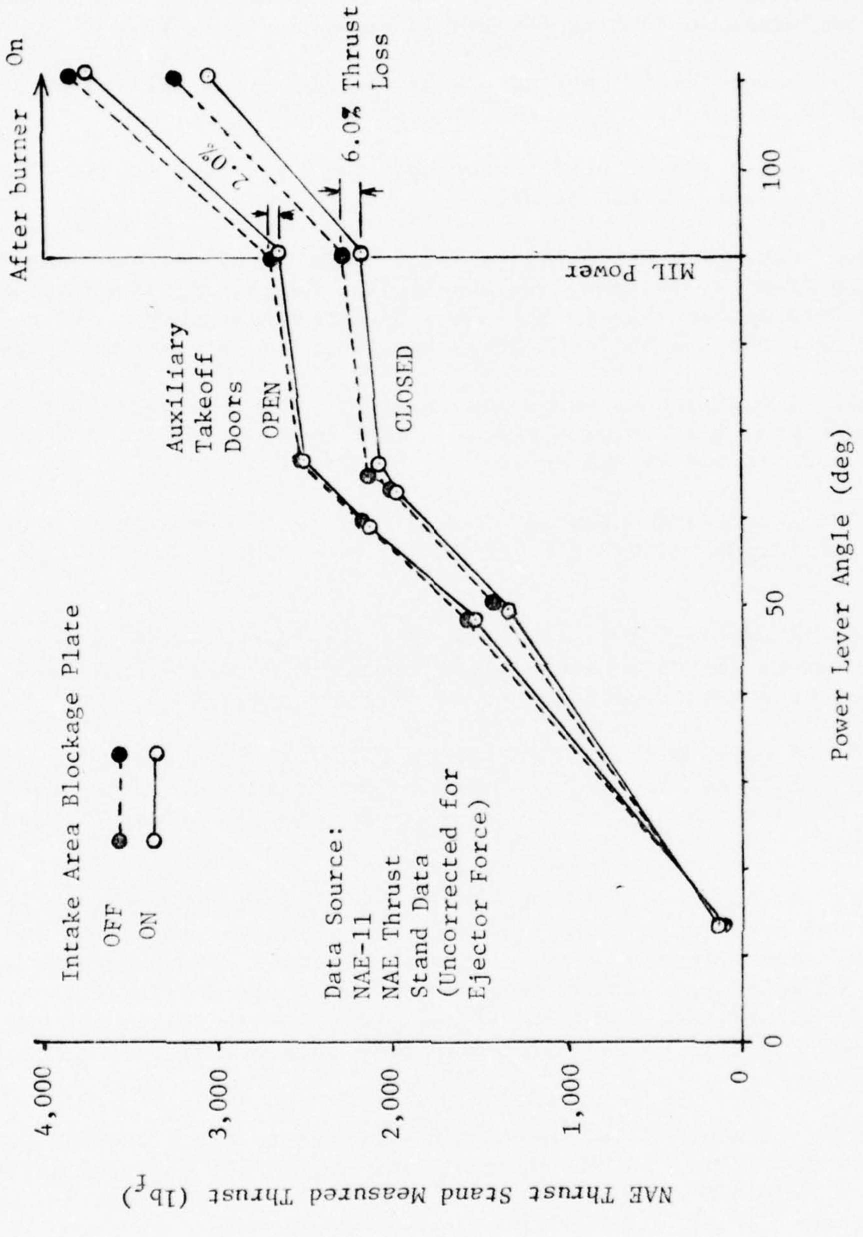


Figure 26: Effect of Intake Area Blockage on NAE Test Stand Thrust

5.2 FLIGHT TESTS

5.2.1 Flight test data for all 11 test flights have been plotted and are presented in Volume III of this report. A brief summary of the various activities performed in testing the TMS is shown in Table V.

5.2.2 FT-01. Level flight testing at 40K and 20K ft as well as engine transient tests at 40K ft. V_{MAX} was reached at 40K ft.

5.2.3 FT-02. Level flight performance data were acquired at 36K, 25K, and 10K ft. V_{MAX} was reached at 36K ft.

5.2.4 FT-03. Maximum power climb to 47K ft. Level flight performance data were acquired at 45K, 20K, 10K and 5K ft. One P_{S6} pressure probe developed a leak during this flight. P_{S6} plumbing was blocked off and flight testing continued but only three P_{S6} pressure probes were in service.

5.2.5 FT-04. A modified Rutowski minimum fuel climb was made from 3K to 45K ft using a prescribed power schedule based upon altitude. Afterburner lights were made at 45K ft and an airstart at 20K ft.

5.2.6 FT-05. A modified Rutowski minimum fuel climb was made from 3K to 30K ft. Level flight performance data were acquired at 30K, 20K and 5K ft.

5.2.7 FT-06. A modified AOI climb was made from 3K to 32K ft and a MAX power Rutowski climb schedule was used from 32K to 38K ft. Level flight performance data were acquired at 30K, 20K and 15K ft.

5.2.8 FT-07. A climb was made from 5K to 30K with the port engine at 95% RPM and test engine at MIL power. Level flight performance data were acquired at 30K and 10K ft. A dive to V_{MAX} was made in a descent from 30K to 20K ft.

5.2.9 FT-08. A climb was made from 5K to 30K with the port engine at 94% RPM and the test engine at MIL power. The climb was continued to 40K ft with both engines at MAX power. This was followed by a push over and dive to 36K ft and V_{MAX}. A MAX power level flight was made at 10K ft. A test was made in order to determine the effect of G-loading on the TMS at 10K ft and 400 knots indicated airspeed. Data were acquired at 1,2,3,4 and 5 G-loadings.

5.2.10 FT-09. A Rutowski MAX power, minimum time to climb profile flight was made from 2K to 34K ft. The schedule was broken off when the aircraft performance diverged from the theoretical plan. A test was made in order to determine the effect of sideslip on the TMS performance by using full left and right rudder at MIL power, 0.33 Mach and 0.7 Mach numbers.

Table V Summary of Flight Test Activities

MANOEUVRE	FT-01	FT-02	FT-03	FT-04	FT-05	FT-06	FT-07	FT-08	FT-09	FT-10	FT-11
Takeoff using (power)	A/B	MIL	A/B	A/B	MIL	MIL	MIL	MIL	A/B	A/B	A/B
Level flt accel. at (power)		MAX	MAX								
MAX power climb to (K ft)	40		47							30	
MIL power climb to (K ft)		36									
Profile type climb to (K ft)				45	30	38	30	40	34		35
Dive MAX power from/to (K ft)	36 20	36 25	30 20	45 36	30 20						
Penetration descent (K ft/K ft)	25/10	45/30								30/20	45/20
Level flt constant Mach at (K ft)	40					30 20	30 10				
Level flt RH engine MAX at (K ft)	20	36 25	45 20		30 20			36 10		30 5	
Level flt RH engine MIL at (K ft)	20	10	20 5		30 5	15		36		30 5	
Level flt at (K ft) RH engine % Reference Thrust			10 85								
MAX power to V_{MAX} (K ft)	40	36									
Engine transient at (K ft)	40		45								
A/B lights at (K ft)				45							45
Airstarts at (K ft)				20							20

5.2.11 FT-10. A MAX power, AOI schedule climb was made from 2K to 30K ft. Level flight data were acquired at 30K, 20K and 5K ft. One P_{S6} pressure probe developed a leak during this flight. It was decided that since the P_{S6} probes were failing at a rate which was not predicted by the PFRT, all four probes would be replaced prior to continuing with flight testing.

5.2.12 FT-11. A Rutowski minimum fuel profile climb was made from 2K to 35K ft. Engine performance tests and afterburner lights were made at 45K. An airstart trial was made at 20K ft.

5.2.13 Tables showing test flight information such as test type, pilot's actions and comments, meteorological data, etc., have been prepared and are included in Volume II .

5.2.14 Photopanel data have been compared with digitally recorded data in order to show consistency of data. Data tables are included in Volume II .

5.2.15 Pilot qualitative evaluations were obtained as a result of four AETE test pilots having made flight tests with the TMS in operation. The pilot's report is included in this volume as Exhibit A.

5.2.16 Data were acquired which permitted studies to be made of the usefulness of a TMS in certain operational roles. A detailed analysis and discussion of these studies is presented in Section 7 of this volume.

5.2.17 Steady state data were acquired at altitudes from 5K to 40K which enabled an analysis of the repeatability of the TMS both in MIL and MAX afterburning power settings and over a speed range of 0.6 to 1.2 Mach number. A detailed discussion and a presentation of these results are made in Section 6 of this volume.

5.2.18 One electronic integrated circuit used in the TMS output to the recorder interface failed during a bench test prior to the aircraft acceptance test for the flight trials. Repair was accomplished without delaying the trials.

5.2.19 The TMS was serviceable at the conclusion of flight testing.

5.2.20 Engine running at the NAE thrust stand, acceptance flights and flight testing resulted in the pressure probes accumulating test lives as follows. Test life time excludes engine running time for trim purposes or ferry flights. Flight time during which a probe failed was not counted.

(a) P_{T5} total pressure probes were all replaced prior to acceptance flight AE-3A. Replacement probes accumulated 10 hr 07 min without failure.

(b) P_{S6} static pressure probes:

(i) 15 hr 14 min to failure of one probe.

(ii) 20 hr 48 min to failure of second probe.

(iii) 21 hr 27 min without failure of two probes. All P_{S6} probes replaced after FT-10.

(iv) 0 hr 40 min without failure of final probe set.

(c) P_{S7} static pressure probes accumulated 22 hr 07 min without failure.

5.2.21 Recorded data were obtained which permitted an analysis of any periodic faults which occured during the flights. The complete analysis is presented in Appendix IX of this report and a discussion of this analysis is made in Section VI.

5.2.22 Flight tests FT-01 to FT-11 provided sufficient data to permit an examination of the dependence of percent reference thrust upon nozzle position. These data are presented in Figure 27.

5.2.23 Flight FT-08 included a test to determine the effect of G-loading on the TMS. A wind-up turn was performed to increase the G-loading from 1-G to 5-G. The test engine was set at MIL power and a target altitude of 10.5K ft and 0.7 Mach number were maintained as well as possible. Pressure corrected gross thrust data, averaged over five second intervals at the test G-loads, have been plotted in Figure 28. Since these data did not correlate well with G-loading, pressure corrected thrust was plotted as a function of Mach number, in Figure 29, and the points annotated with respect to G-loads. Pressure corrected reference gross thrust data were included in Figure 29 to further substantiate conclusions. Data reduction print-outs indicate that the complete TMS operated without fault during this test.

5.2.24 Part of FT-09 included a test to determine the effect of sideslip on the TMS. The QTP maintained altitude and Mach number while recording a reference point and the points at full left and right rudder. This test was performed both at Mach 0.33 and 0.70. The resulting thrust data were pressure corrected to account for any small altitude change and plotted against Mach number in Figure 30.

5.2.25 Rutowski profile climbs were attempted to gain experience with the TMS in an operational role. Thrust, aircraft drag and predicted meteorological data were provided to the USAF. Rutowski profiles were generated by the USAF for the CF-5D aircraft. The QTP prepared for the Rutowski profile climbs by making practice flights in another CF-5 aircraft.

5.2.26 The data flight, FT-09, was made on a day in which the ambient temperature was approximately $40^{o}R$ warmer than the data for which the profile was generated. The resulting flight path followed the Rutowski profile from 2K ft to a target 34K ft and through a MAX A/B supersonic dive to 30.4K ft. This was to be followed by a supersonic climb to a target of 36K ft. The aircraft reached 34K ft taking only ½ second longer than predicted. Mach numbers were well matched to the profile. The aircraft Mach number was 1.155 rather than the target 1.148 at the end of the supersonic dive but a target Mach 1.20 could not be achieved during the final climb. The mission was aborted at 34K ft and Mach 1.15.

5.2.27 A minumum fuel Rutowski climb profile was attempted as part of flight FT-11. The aircraft climbed from 2K to 30K ft using MIL power but required two minutes longer than was expected. Some 26% (100 lb) more fuel was required than was predicted for this portion of the climb. Minimum A/B power was used in climbing to 33.5K ft and MAX A/B was used in climbing to 42K ft. The target Mach number for this peak altitude was 0.974 and the actual value was 0.973. The test profile remained approximately 2 minutes late. A MAX A/B dive was intended to gain Mach 1.2 at 36K ft but only Mach 1.13 was achieved. Fuel consumption exceeded the predicted profile total consumption by approximately 150 lb. It is assumed that better results were not achieved in part because the predicted meteorological data did not match the actual values. As a result, the aircraft excess thrust was lower than had been predicted.

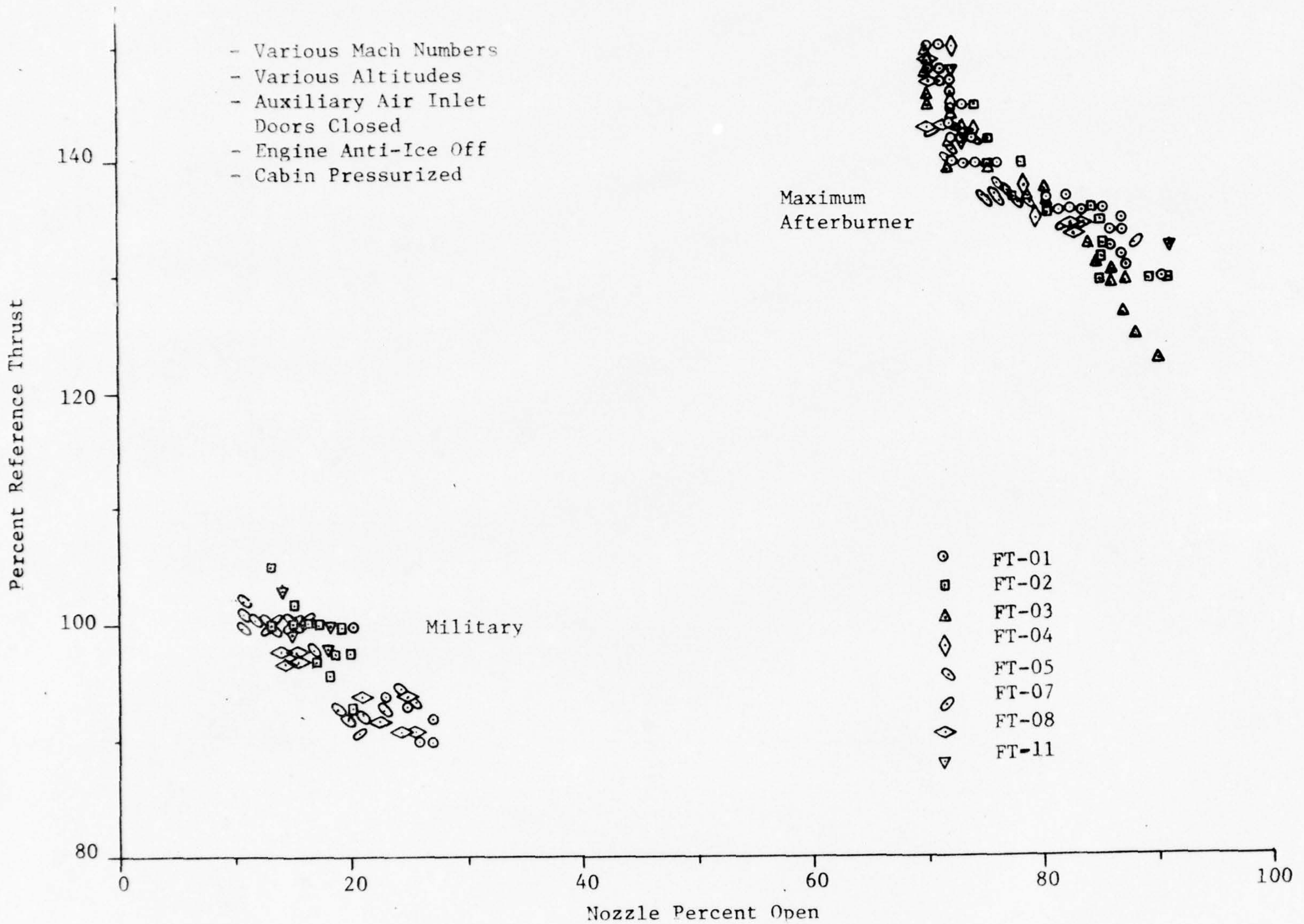


Figure 27: Percent Reference Thrust vs Nozzle Position

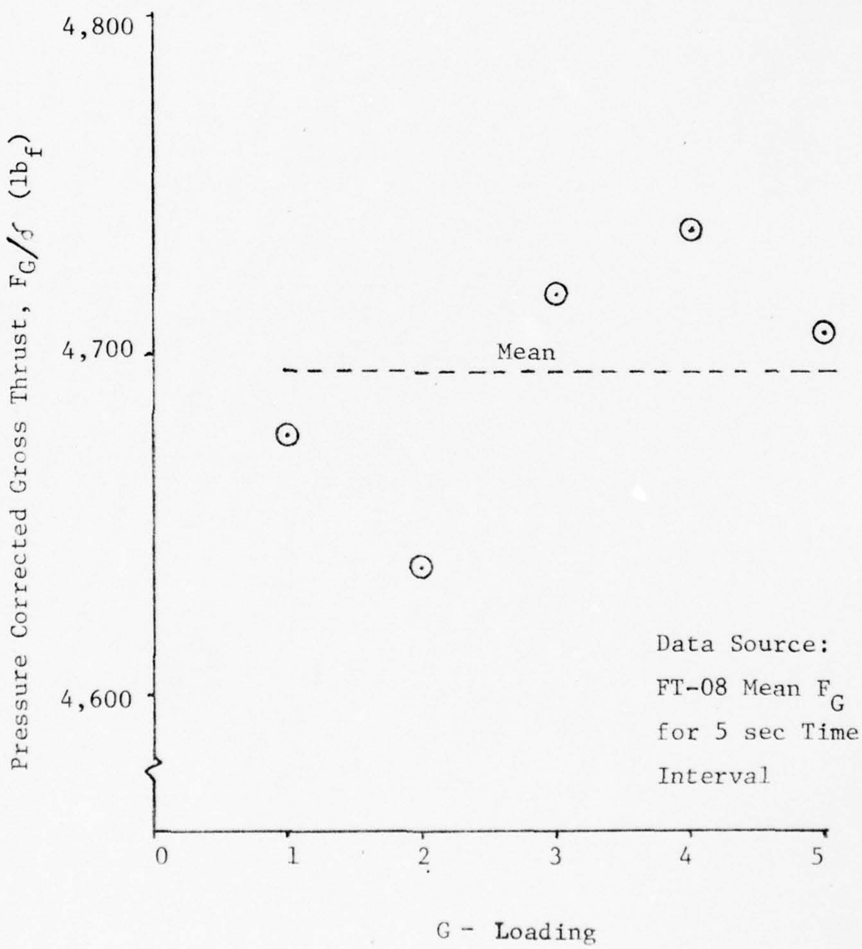


Figure 28 Pressure Corrected Gross Thrust vs G-Loading

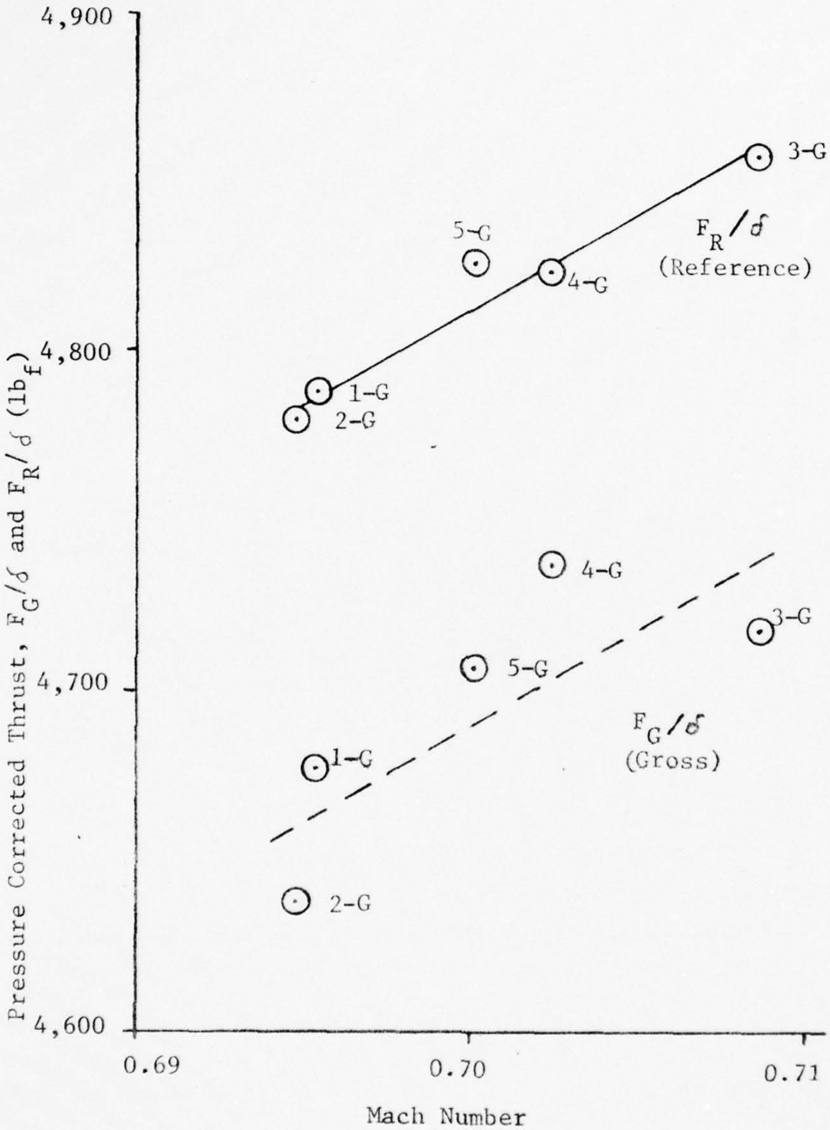


Figure 29 Pressure Corrected Thrust vs Mach Number

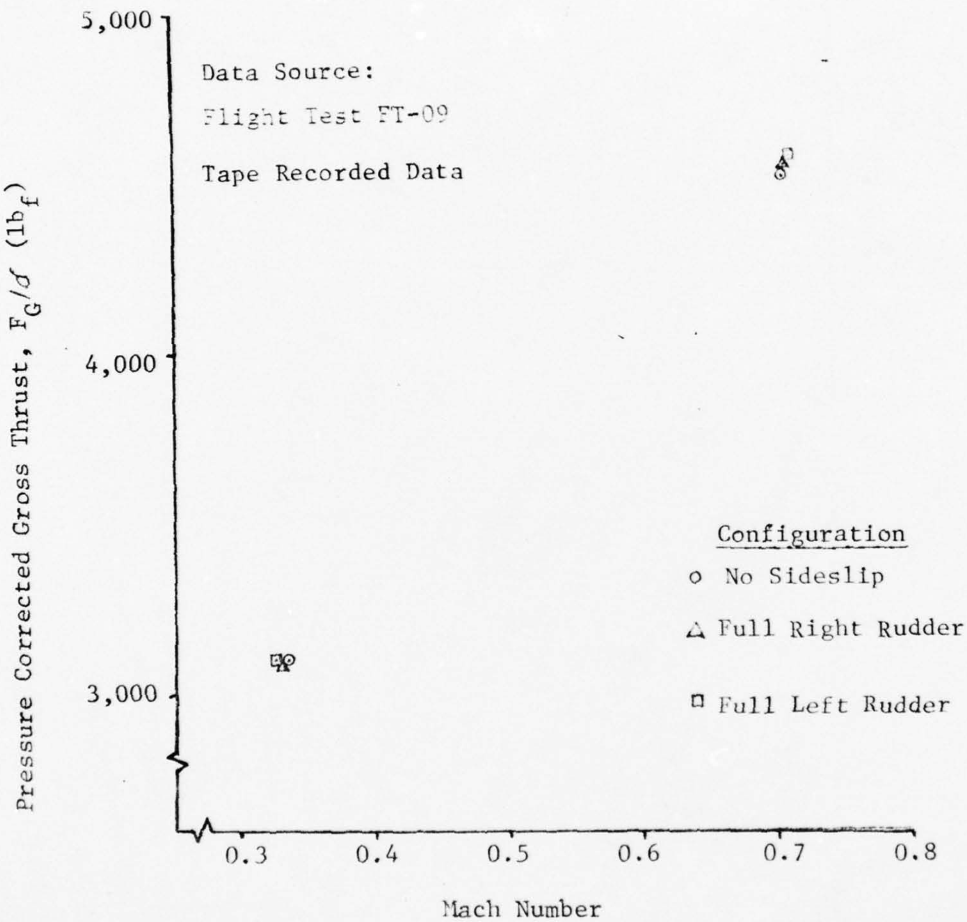


Figure 30 Effect of Sideslip on Thrust Measuring System

SECTION VI

ANALYSIS AND DISCUSSION OF RESULTS

6.1 GROUND TESTS

6.1.1 Bare Engine Cell

6.1.1.1 Pressure probes were designed such that installation and removal from a test engine was not difficult with the engine removed from the aircraft. The PFRT served to prove the flight safety of the probes and the satisfactory locations for the probes and plumbing.

6.1.1.2 A need for delay volumes or other means of synchronizing pressure data was demonstrated by bare engine running.

6.1.1.3 Bare engine running proved that the gross thrust equation could be calibrated to a particular engine. The fact that two engine tailpipes were instrumented and thrust was computed with equal success indicates that the equation calibration is probably valid for other individual engines of the same type.

6.1.1.4 Tracking and repeatability of the TMS is displayed in Figures 17 and 18. Two of the 122 points shown in Figure 17 fall outside of a $\pm 2\%$ F.S. deviation band. Deviation data show no tendency to depart from the line of perfect agreement over the range 368 to 4171 lb_f. Repeatability is assured by the fact that data have been acquired from 16 distinct test runs made over a time span of two years.

6.1.1.5 Dynamic engine tests have shown that the TMS will respond to rapid engine thrust changes. Figures 19 and 20 show that tracking time lags are of such short durations that they may not be discernable by a pilot.

6.1.1.6 It has been shown that practical maintenance test tools can be designed which will aid in making engine probe leak tests. Leak checks may now be made with the J85-CAN-15 engine installed in a CF-5D aircraft.

6.1.2 Static Thrust Stand Tests

6.1.2.1 A method of accounting for ejector force in order to compute gross thrust from static aircraft thrust stand indicated thrust data is presented in Appendix V.

6.1.2.2 An extensive data set has been recorded and is presented in Volume II .

6.1.2.3 The TMS is a good indicator of loss of engine performance due to duct loss. Duct losses are reduced by opening the auxiliary takeoff doors. The effect of opening these doors was clearly demonstrated by test stand measured and TMS indicated thrust increases in the order of 15% at MIL power.

6.1.2.4 Tracking and repeatability of the TMS on the NAE test stand is shown in Figure 21 for the pre-flight test trials. Two of the 203 data points shown in Figure 21 fall outside of the $\pm$2% F.S. bands. Tracking remained as good as it was for the bare engine tests. Run NAE 8 data for the SE transducer test produced data which correlated with the thrust stand data.

6.1.2.5 The fact that the TMS continued to operate without apparent degradation after the P_{T5} probe failure serves to provide a measure of the fail-safe capability of the system. That is, some leakage can be tolerated by the TMS without the system falsely indicating an engine failure. Some further test cell trials could be conducted to determine the extent of damage the system can sustain and still operate accurately. This point would be of particular interest to military personnel considering battle damage effects on the TMS.

6.1.2.6 Post flight-test static trials indicated that the TMS tracking and repeatability characteristics had not deteriorated. In fact, Figure 22 shows a closer agreement between the TMS and the test stand. One of the 145 data points (Figure 22), exceeded the ±2% F.S. deviation band. This result is attributed to the use of S.E. transducers.

6.1.2.7 Dynamic responses of the TMS, shown in Figures 23 and 24, remained satisfactory to the conclusion of the trials. The worst case was a one second lag.

6.1.2.8 The P_{S7} static probes operated for 28 hr 10 min of flight and ground testing without failure. This probe design should be used as a guide in designing future probes. The P_{T5} probe failure and the two P_{S6} probe failures indicate the need for further design work in order to develop satisfactory probes for these engine locations. Since the PFRT led to a false assumption that the probes would remain structurally secure through a flight test, it must be assumed that the PFRT did not completely represent flight test conditions. It is quite probable that the flight test G-loadings contributed to probe failures. Also, in-flight engine operating conditions such as engine temperature, time in A/B, temperature variations, or vibrations may have been more severe than those of the PFRT.

6.1.2.9 Inlet air duct blockage will result in a gross thrust loss. The present cockpit instrumentation will not indicate intake conditions other than the auxiliary takeoff door position. An intake blockage plate was used to simulate an intake air duct blockage and thus demonstrate the

effect of an air blockage on engine thrust, see Figure 25. Table IV lists cockpit engine instrument and TMS indications for a number of tests made with and without intake blockage. Note that with intake blockage for the MIL power case with the auxiliary takeoff doors open, RPM remained at 100%, the nozzle changed from 18.9% to 18.5%, fuel flow decreased by 50 lb/hr (a change of 1.6%) but the TMS indicated a percent reference thrust loss of 2.9%. Adding the intake blockage plate with the auxiliary takeoff doors closed resulted in a nozzle position change from 20.3% to 22%. Fuel flow changed by 85 lb/hr (422-337), or a change of 2.7% but the percent reference gross thrust changed from 82.5% to 76.3%, or a change of 6.2%. Although the NPI and fuel flow indications changed with intake blockage, the indicated values remained within normal operating limits. No abnormality would be immediately noticed by the pilot. The TMS will indicate a thrust loss due to an air blockage upstream of the compressor face. Since an intake blockage could be caused by structural damage, icing, bird ingestion, etc., it is important that a thrust loss of this type be detectable.

6.1.2.10 Opening the auxiliary takeoff doors while the aircraft is stationary and the engine is operating at MIL power causes an increase in power of about 14% to 16%. This power increase is indicated by both the TMS gross and percent reference thrust indicators. A pre-takeoff engine check could be made by observing the percent reference thrust at MIL power. The TMS should indicate approximately 100%. An indication of approximately 85% would remind the pilot that the auxiliary takeoff doors had not been opened. An examination of thrust data acquired only during actual pre-takeoff and post-landing static aircraft engine checks produced the mean data of Table VI. These data show slightly smaller changes than the 103 data points from the NAE trials which are shown in Table IV. Table VI lists the mean thrust changes due to opening the auxiliary takeoff doors, selecting anti-icing on or selecting cabin pressure. Appropriate pre-takeoff check values would be obtained with sufficient experience with using the TMS as a diagnostic aid.

Table VI: Mean Gross and Percent Reference Thrust Change Due to Selecting Various Controls

	Auxiliary Doors Open	Anti-ice On	Cabin Pressure On
Mean Change			
Gross Thrust	+ 411 lb	- 138 lb	- 100 lb
Reference Thrust	13.9%	- 4.6%	- 3.4%
No. of Sample Points	23	3	4

6.1.2.11 Tracking and repeatability capabilities of the TMS are shown in Figures 21 and 22. The TMS is able to track the engine thrust over the range of 120 pounds to 4160 pounds. There is no obvious tendency for the mean TMS indication to depart from the line of perfect agreement at any point in this thrust range. Approximately 2% of the data points on Figure 21 exceed $\pm 2\%$ F.S. deviation. These data include points obtained with Conrac and SE transducers, auxiliary takeoff doors open and closed, and with the inlet area blockage plate on and off. Figure 22 shows an improvement in TMS tracking capability. In this case, one point in 145 is outside of the $\pm 2\%$ F.S. deviation band. One difference between the tests for Figures 21 and 22 is that Conrac transducers were used for almost all of the data points in the first case and SE transducers were used in the latter case.

6.2 FLIGHT TESTS

6.2.1 TMS Performance

6.2.1.1 The TMS was tested over the flight envelope of the test aircraft. V_{MAX} was achieved both in level flight and in dives. The aircraft was taken to a pressure altitude of 47K ft and testing included many performance type manoeuvres. Magnetic data recording enabled the acquisition of sufficient data to permit an analysis of the TMS performance throughout the trials. Many anomalies were detected by means of the recorded data. These are discussed in paragraph 6.2.3.

6.2.1.2 Two P_{S6} pressure probes failed during flight testing. This problem is discussed in paragraph 6.1.2.8.

6.2.1.3 It was noted that deviations from 100% of reference gross thrusts were observed for MIL power operations at various flight conditions. As can be seen in Figure 27, this deviation appears to be a function of nozzle position. As noted in paragraph 5.2.22 there were deviations in nozzle area from that predicted by the engine status deck which was used to generate the reference thrust computation, the status deck values having a smaller spread. As thrust is a function of nozzle position it would be expected that the computed gross thrusts would differ from that predicted by the reference thrust computation and cause deviations in reference thrust from 100%. A similar situation prevails at MAX power operation. This nozzle behaviour is indicative of a difference in the lapse rates between the test engine and the average engine status deck data used to generate the percent reference gross thrust. (Lapse rate is defined in Appendix VII.)

6.2.1.4 TMS operation was not interrupted during the G-loading testing to 5-G. No thrust deviation could be detected and related directly to G-loading.

6.2.1.5 Indicated gross thrust, when corrected for ambient pressure differences, did not indicate a significant change in thrust as a function of sideslip at Mach number 0.33 or 0.70 as is indicated by Figure 30.

6.2.1.6 Rutowski profiles are based upon a prior assumption about ambient conditions and aircraft excess power. If the basic data are in error, then the profile becomes impossible to achieve. The test day ambient temperatures at altitude were approximately 40°F above predicted temperatures. Therefore, the aircraft engine could not achieve predicted thrust levels. The Rutowski profiles of FT-09 and FT-11 had to be aborted when it became obvious that this portion of the mission could not be successful. This suggests the need for an on-board computer to provide real-time up-date information to the pilot.

6.2.2 Consistency and Repeatability

6.2.2.1 Consistency and repeatability have been proven by an extensive analysis of flight data. A complete description of the analysis is presented in Appendix VIII. This analysis includes an examination of 200 data points from flights FT-01 to FT-11 and at altitudes from 5K to 45K ft. It shows that the TMS computed thrust can be presented as a thrust parameter and correlated to Mach number over a Mach range of 0.6 to 1.2. Since these data collapse to single lines for all reference altitudes and within an approximate deviation of 1%, TMS consistency is clearly shown. This analysis does not imply a thrust precision. It has been shown that the actual flight test gross thrust at MIL power does not agree with the TMS stored reference equation. It has also been shown that the TMS's method of reference thrust calculation does agree (up to 35K feet) with the average engine status deck data modified to account for installation effects according to manufacturer's data. The analysis does not provide proof that either the gross thrust equation or the reference thrust equation is in error but only that they do differ at altitude. Information in Appendices VI and VII indicate the complexity of the engine lapse rates problem and associated reference thrust situation.

6.2.3 Anomalous Data

6.2.3.1 An extensive diagnosis has been made of all the data anomalies recorded during all of the test flights. Data were recorded at ½ second intervals and computer generated plots aided in analyzing problems. This enabled an examination of many anomalous data points. A complete discussion of the result of this diagnosis is presented in Appendix IX. The following problems are considered to be responsible for anomalies attributable to the TMS operation.

(a) The gross thrust computation is not accurate at very low thrust levels (below 5% reference thrust). A computer overflow causes the gross thrust computed value to increase from about 170 to 1440 lb_f. This is caused by operating the system at thrust levels below its design limit.

(b) Gross thrust computation errors were observed during engine transients. It is believed that the delay volumes do not synchronize the engine pressure data completely and erroneous computations result from using unsatisfactory data.

(c) Reference thrust was erroneously computed as zero on eight brief occasions. The cause of this remains undefined.

(d) An electrical transient is believed to have caused the reference computer to output erroneous data on five occasions but only for single ½ second updates.

(e) A design problem in the reference thrust indicator resulted in one indicator pointer problem such that the pointer indicated 160% throughout one flight.

(f) Gross thrust was computed as a value which erroneously fluctuated by 300 to 400 pounds during three flights. The problem was traced to pressure transducer data being input to the computer. The problem could have originated in the transducer, the A/D converter or the connecting circuit. Post-flight static transducer tests did not detect transducer faults.

(g) The reference thrust indicator lags the engine thrust during engine transients. On throttle advance the lag is 0.65 sec while on throttle retarding the lag is 0.82 sec. This is a design limitation which can be overcome.

6.2.4 Pilot Observations and Comments

6.2.4.1 A summary of pilot observations and comments is contained in the letter of 14 Sept 73 signed by Colonel L. H. Keelan, Commander, AETE, which is attached as Exhibit A to this volume. Some of the pilot's comments include the following:

(a) The scale of the present indicator allows misreading of the percent reference thrust.

(b) The indicated thrust value followed throttle movement with little apparent lag.

(c) Percent reference thrust increased approximately 14% when the auxiliary takeoff doors were opened at MIL power.

(d) Percent reference thrust remained near 100% at MIL power throughout the flight envelope.

(e) The TMS does not positively indicate in-flight engine shutdown or relight.

(f) "All four test pilots agreed that a thrustmeter would be valuable during an engine check just prior to takeoff. They definitely desired some indication of engine thrust for this check. One pilot with considerable EPR (Exhaust Pressure Ratio) gauge experience believed an EPR gauge gave equally valuable information. The other three pilots, two with considerable EPR gauge experience, strongly preferred a thrustmeter."

(g) "During test or functional check flights the engine operation can be rapidly checked throughout the flight envelope, dealing with unsubstantiated pilot comments such as a particular aircraft 'felt underpowered'. Modifications to aircraft engines can be easily assessed inflight with regard to their affect on thrust output. Drag measurements on new aircraft or new aircraft configurations, including external stores, can be rapidly carried out throughout the flight envelope."

(h) "For an operational pilot during flight, the raw data displayed on the thrustmeter would appear to be of use only as an aid in selecting cruise, climb, or descent power settings. However, if the thrustmeter data are converted to such information as thrust-to-weight ratio, excess thrust (thrust minus drag), or specific energy, it may be possible to devise some kind of display which would show a pilot an optimum manner in which to manoeuvre his aircraft."

6.3. PRE AND POST FLIGHT TEST EQUIPMENT CALIBRATIONS

6.3.1. Systems used to measure aircraft variables were calibrated by AETE. Calibrations were repeated regularly. Calibration data are included in a special error analysis report presented in this volume as Exhibit B.

6.3.2 S.E. transducers and Conrac transducers were calibrated before being used. The S.E. transducers were calibrated in Apr 73 following the final engine testing. Transducer calibration data are presented in Volume II .

6.3.3 The ComDev TMS computer and indicators were calibrated in Feb 73. Calibration data are in Volume II .

6.3.4 The pre-amplifier for the Rosemount probe, the converter used in indicating aircraft engine RPM, and the CADC were calibrated in Feb 73. ComDev specified modifications only were calibrated in the CADC. Calibration data are in Volume II .

6.4 EQUIPMENT RELIABILITY

6.4.1 The thrust measuring computer and an indicator have been subjected to environmental and electromagnetic interference testing. They were subjected to temperature, altitude, humidity, vibration and shock testing. Magnetic, transient and radiated susceptability tests along with radiated magnetic field measurements were also carried out on the system.

6.4.2 S.E. Laboratories (Engineering) Ltd. specify that their S.E. 40 series transducer production units will have a guaranteed minimum operational life in excess of 56,940 hours Mean Time Between Failure (MTBF). It is planned that three S.E. transducers will be extensively tested in the ComDev laboratory in order to further assess their applicability to the TMS.

6.4.3 The present TMS computer design is a compromise between available parts and parts which could be specified if a long MTBF is desired. Therefore, early MTBF study data are not applicable to the prototype. Since the completion of environmental and electromagnetic interface testing, the computer system operated for 698 hours without a component failure. One electronic integrated circuit in the TMS interface output to the data recorder failed during bench testing in Dec 1973. This circuit would not be required except for the need of a magnetic tape recorder for data acquisition. The TMS was serviceable at the conclusion of the flight and ground running trials.

6.4.4 One indicator design problem (pointer indicated 160% throughout one flight) was encountered during flight trials. The problem is explained in Appendix IX and can be eliminated in future indicators.

6.4.5 Engine probe life must be extended by designing more reliable probes. The achieved life has been detailed in Section 5. Since EPR probes are in use with many engines, it should be possible to design reliable probes for a TMS. Diffuser strut movement relative to the engine casing made P_{T5} probe design difficult. The narrow space between the afterburner liner and the casing made P_{S6} design difficult.

Clearance between the VEN and engine casing made P_{S7} design difficult in the original a position. Using the b position seems to have made it possible to design suitable P_{S7} probes.

6.5 PROBABLE ERROR OF DATA ACQUISITION SYSTEM

6.5.1 The data acquisition system, including the sensors and recorder, have been examined and an estimate of the probable errors in acquiring, recording and computing data has been made. A complete report on the study is presented in this volume as Exhibit B. While all of the recorded data are of interest, only data pertaining to the operation of the TMS need be considered here. Exhibit B reported probable errors due to hysteresis, manufacturer's tolerance, calibration, A/D conversion and truncation of data for recording purposes. Table VII lists the probable errors in acquiring and recording TMS data.

Table VII: Probable Errors in Acquiring and Recording TMS Data

Variable	Probable Error
Ambient air temperature (^oR)	$\pm 1.5^oR$
Mach number ($0.17 \leq M \leq 1.6$)	± 0.012
Ambient static pressure (psia)	± 0.077 psia
$P_{S6} - P_{S7}$ (psid)	± 0.081 psid
$P_{T5} - P_{S6}$ (psid)	± 0.054 psid
P_{S7} (psia)	± 0.32 psia
Recording gross thrust (lb)	± 2 lb
Recording % ref. thrust (%)	$\pm 0.02\%$

SECTION VII

OPERATIONAL POTENTIAL

7.1 POWER CHECK PRIOR TO TAKEOFF

7.1.1 Engine Thrust Check

7.1.1.1 A percent reference thrust indication of 100% at MIL power indicates that the engine is operating as an average engine as specified by the manufacturer. Operational engines will display deviations from average operation resulting in deviations from the 100% reference thrust indication. It should prove possible to specify a range of percent reference thrust within which the engine performance is considered acceptable. The percent reference thrust indicator will then serve as an engine monitor and may be used as a power check instrument prior to takeoff.

7.1.1.2 Opening the auxiliary takeoff doors with a static aircraft and MIL power engine causes an increase in indicated gross thrust of about 411 lb or about 15% reference thrust. Gross thrust decreases of about 138 lb were observed when the anti-icing was selected ON or about 100 lb when the cabin pressure was selected ON. This suggests that the pilot could make a pre-takeoff engine check by observing the percent reference thrust at MIL power. An indication of approximately 100% should be observed. An indication of approximately 85% would remind the pilot that the auxiliary takeoff doors had not been opened.

7.1.2 Gross Thrust Indicator as a Takeoff Aid

7.1.2.1 A gross thrust measuring system may be useful as a pre-takeoff check instrument. Present takeoff charts are based upon predicted engine performance and may not necessarily represent the actual engine thrust due to trim changes or ambient conditions. Engine actual gross thrust could be used in predicting ground roll and air run distances. Air run distances are usually defined as the horizontal distance between the takeoff point and the point at which the aircraft would clear a 50 ft obstacle. A pilot could read a gross thrust indicator and know the actual engine thrust available just prior to brake release. Non-standard runway condition corrections could be made by means of a chart or pocket computer. Alternatively, a pilot may choose to use a chart and determine a minimum safe gross thrust or safe percent reference thrust for the takeoff conditions. In either case, the ground roll and air run distances or the balanced field length (critical field length) would be known for the actual gross thrust available. The TMS would then be a GO-NO-GO indicator.

7.1.2.2 The pilot did not attempt to make maximum performance takeoffs and climbs to 50 ft as required in acquiring data for AOI purposes. Theodolite data were not acquired during the test. However, the above theory has been tested by making reference to takeoff data from 11 test flights. The pilot used an event code to mark the recorded data at the time of brake release. The CADC static

pressure data were used in determining the point at which the aircraft had moved through 50 ft above the brake release pressure altitude. The time intervals between this point and brake release were plotted in Figure 31 against gross thrust at brake release. Data were not corrected for non-standard conditions. A good correlation is shown for the five MIL power cases. Afterburning reduces the takeoff time and all the afterburning cases fall below the MIL power curve. Again, a good agreement is indicated in four of the cases where afterburning was used for 17 seconds. (The pilot was able to achieve a wide range of MIL powers at takeoff by manipulating the auxiliary takeoff doors, anti-icing and cabin pressure).

7.1.2.3 Estimated distances required from brake release to the end of the air run were made by integrating the net thrust force over the time interval. Net accelerating thrust was obtained by subtracting ram drag, friction drag and airframe drag from gross thrust. The resulting net thrust was resolved as a function of time and equated to the product of aircraft mass and acceleration. True speeds at the end of the runs were used in the integration in order to improve the solution. AOI data are based upon a MIL power ground roll and an afterburner air run.

7.1.2.4 The above analysis serves only to demonstrate the feasibility of using thrust data in determining takeoff performance. In practice, the analysis would have to consider all the normal corrections for non-standard conditions.

7.1.2.5 From the above analysis, it appears that the gross thrust data may be correlated to ground roll and air run distances. Any use of afterburning will reduce these distances. A gross thrust indicator would present the pilot with an actual available thrust figure for use in predicting takeoff data.

7.2 CONTROL VARIABLE FOR VARIABLE THROTTLE ENERGY MANAGEMENT MISSIONS

7.2.1 The Rutowski minimum fuel climb profile includes a climb using the power lever to control afterburning. As a result of controlling afterburning, thrust is managed. Therefore, this section of the profile could be defined such that the pilot would use the power lever and control thrust directly by observing achieved thrust as indicated by a thrustmeter. Since thrust is related directly to fuel flow, the desired minimum fuel consumption will follow the target thrust. Thus, the minimum fuel climb should be achieved if all other specifications are met.

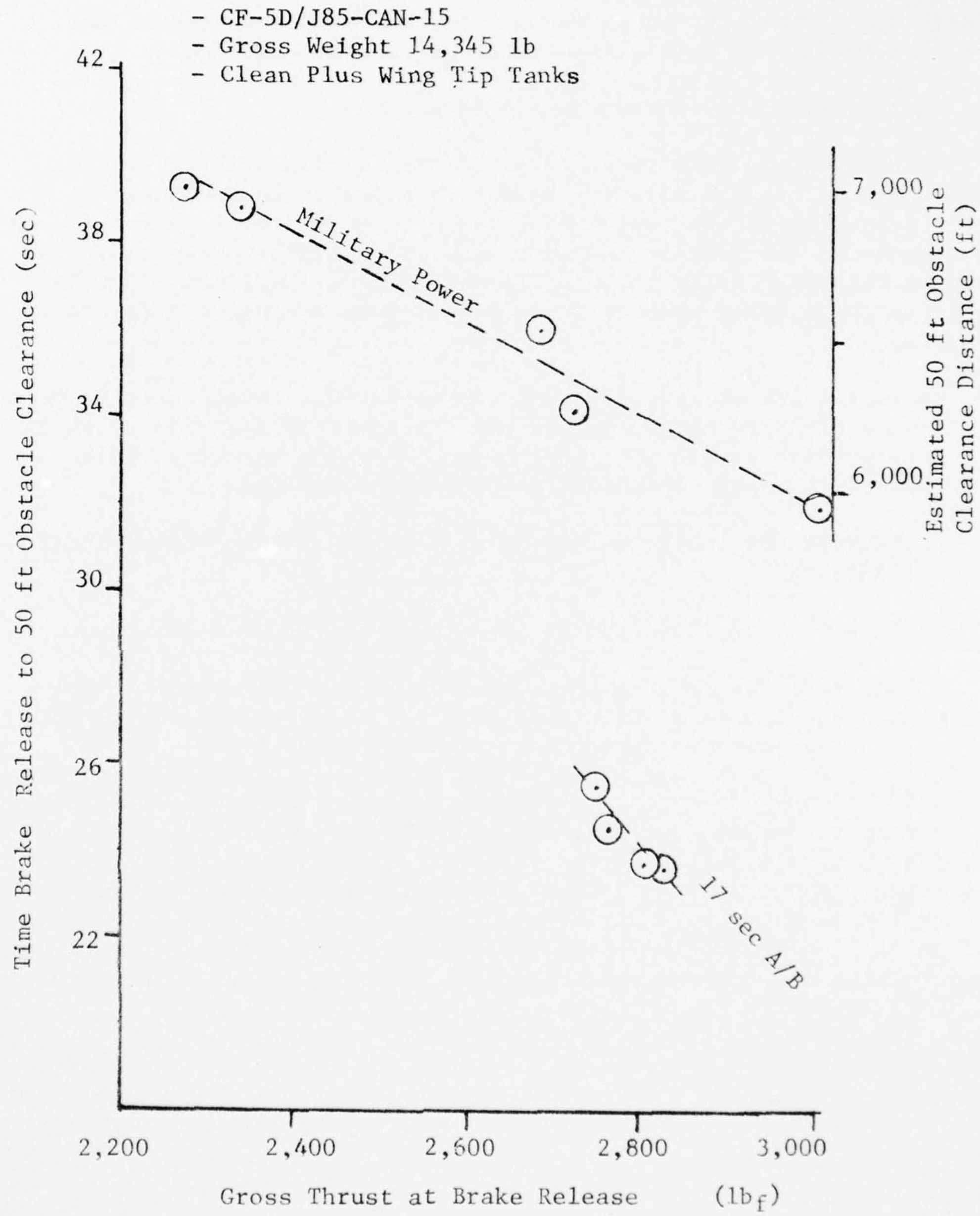


Figure 31 Takeoff Time vs. Indicated Gross Thrust

AD-A040 092 COMPUTING DEVICES CO OTTAWA (ONTARIO) F/G 21/5
EVALUATION OF AN AIRBORNE THRUST COMPUTING SYSTEM. VOLUME I. SY--ETC(U)
MAY 75 J A GRAVELLE F33657-69-C-0733
UNCLASSIFIED H036/119/FR/I ASD-TR-75-2-VOL-1 NL

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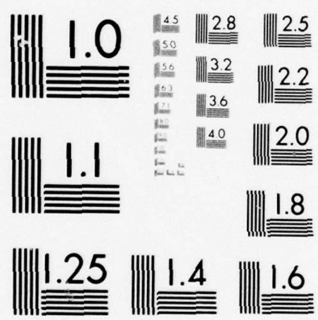

MICROCOPY RESOLUTION TEST CHART
NATIONAL BUREAU OF STANDARDS-1963-A

7.2.2 Energy manoeuverability concepts have been used in defining many optimum flight paths. The thrust measuring system could probably become an asset to the pilot by providing actual thrust values for the engine performance rather than predicted values. These actual thrust values when fed into an on-board computer may play an important role in the design of automatic or semi-automatic flight control systems.

7.3 TRIM VARIABLE FOR ENGINE MAINTENANCE

7.3.1 A TMS could be used in performing line maintenance trim checks. The TMS would be used to trim the engine with the engine installed. A specific fuel consumption computation could be made directly from the TMS indicated thrust and cockpit indicated fuel flow. This would establish the engine rejection criteria with the aircraft on the line. This line maintenance check would warn of the need for more extensive maintenance procedures.

7.3.2 Using the TMS as an additional diagnostic tool would help in more accurately determining engine health and the possible need for an engine change. This could result in a time and cost saving by reducing the present number of engine changes for maintenance reasons.

7.4 INSTRUMENT FOR DETERMINATION OF DRAG DUE TO AIRCRAFT CONFIGURATION ALTERATIONS

7.4.1 Incremental Drag Coefficient Due to Configuration Alterations

7.4.1.1 Indicated TMS gross thrust data may be used to determine the incremental drag change due to alterations in the aircraft configuration. This fact was demonstrated during part of flight test FT-09. An analysis of flight test data acquired for this demonstration is presented in Appendix X. This analysis determined the incremental drag coefficients, ΔC_D, due to extending landing gear, flaps and speed brakes. Data were resolved both directly from thrust readings and from thrust readings plus computed ram drag data, F_{RDG}. The latter results are shown in Table VIII. Note that the drag coefficients for individual configuration alterations correlate very well with the test value for the corresponding configuration where more than one configuration alteration has been made at the same time.

7.4.1.2 Improved ΔC_D data are attained if F_{RDG} differences are computed.

Table VIII: ΔC_D Due to Configuration Alterations

CONFIGURATION	F_G (lb)	F_{RDG} (lb)	M	P_{SO}	ΔC_D
1. CLEAN	1189.0	419.5	0.3829	12.20	-
2. SPEED BRAKES	1883.7	478.8	0.3824	12.22	0.020
3. LEADING EDGE FLAPS	1091.3	405.0	0.3823	12.21	-0.0026
4. FULL FLAPS	1450.0	437.0	0.3832	12.17	0.0078
5. LANDING GEAR	2253.7	493.4	0.3830	12.23	0.032
6. SPEED BRAKES PLUS FULL FLAPS	2078.6	389.4	0.3816	12.20	0.027
7. SPEED BRAKES + LANDING GEAR	2828.0	488.3	0.3755	12.21	0.053
8. FULL FLAPS + LANDING GEAR	2473.3	494.3	0.3831	12.21	0.039

7.4.2 TMS Used to Obtain Drag Data

7.4.2.1 A TMS can be used in acquiring thrust data which may be used in computing aircraft drag coefficients. This application has been illustrated in Appendix X. TMS gross thrust data were used along with other recorded data and aircraft drag coefficients were computed from the resulting excess power curve.

7.4.2.2 The TMS should be particularly useful in flight testing which involves simultaneous alterations of engine variables and airframe configuration. Gross thrust would indicate the effect of the engine variables. Computed net thrust would equal the aircraft total drag in stabilized flight. Therefore, both engine and airframe testing could be performed during one flight.

SECTION VIII

CONCLUSIONS AND RECOMMENDATIONS

8.1 CONCLUSIONS

8.1.1 Test hardware has demonstrated that the system can be applied to a variable nozzle afterburning turbojet engine for operational use.

8.1.2 Accuracy of the TMS in the J85/CAN-15 test engine has been proven to be satisfactory by an extensive series of bare and installed engine tests using the static thrust measuring stand. TMS gross thrust indications were consistently within $\pm$ 86 lb of the test stand indication. This figure is 2% of maximum, sea level, standard day thrust of a J85-CAN-15 engine.

8.1.3 Tracking capabilities of the TMS have been demonstrated by comparing TMS computed thrust with test stand measured thrust over the range of 300 to 4000 lb. The TMS shows no sign of deviating from agreement with the test stand. TMS computed thrust changes to agree with engine power during dynamic engine operations with an acceptable time lag. The percent reference thrust indication has an inherent lag which may be improved in future designs.

8.1.4 Repeatability and consistency capabilities of the ComDev TMS have been flight tested in a CF-5D aircraft and shown to be satisfactory for the test engine. Gross thrust was repeated within 1% of the point at altitudes from 5,000 to 45,000 ft. Both MIL power and full afterburner power data for a Mach range of 0.6 to 1.2 were used in demonstrating TMS repeatability and consistency.

8.1.5 System calibration remained satisfactory and the constants were unchanged between bare and installed engines, between two engines of the same model and between engines trimmed under various ambient conditions. The calibration was also valid when tail pipes were exchanged between engines. Calibration for the J85-CAN-15 engine was accomplished from the 89 data points acquired during 9 hr 57 min of engine running. (Engine running was accomplished over a period of many months during which time engine probes were being designed and tested). A further 5 hr 29 min of bare engine running produced 33 more data points which confirmed the validity of the calibration constants.

8.1.6. The TMS operated without fault that could be attributed to such flight manoervres as G-loading to 5-G, climbing to maximum test altitude, accelerations to V_{MAX}, subsonic or supersonic flight, or afterburner operation. Following in-flight engine shut down and relight, the TMS continued to operate without fault. In-flight engine slams or chops often affected the gross thrust computer for one or two ½ second computer updates. Delay volumes or some electronic time delaying could be used to alleviate this problem.

8.1.7 The percent reference thrust indication did not match the flight test gross thrust at MIL power at all altitudes.

8.1.8 The reference thrust indicator lags the actual thrust during dynamic engine operations such as a slam or chop. The present lag factor of 0.65 second on a throttle slam and 0.82 second on a throttle chop could have been reduced had this been a design criteria. Future indicators will be more responsive than the present one.

8.1.9 Flight recorded data indicated that fluctuations in computed gross thrust were being periodically output to the indicator during airborne testing. The trouble source could only be traced to the A/D converter, transducer and wiring system. Post flight static tests did not detect a fault within the transducers. Further bench testing should be done in order to clear this problem.

8.1.10 Pilot comments regarding the indicator face markings should be considered in future designs.

8.1.11 The TMS was tested as an indicator of engine performance degradation due to intake air duct blockage. The TMS indicated a gross thrust loss when an intake air duct blockage was simulated. This implies that the TMS can be a valuable aid as an instrument to indicate inlet problem such as those caused by icing and the ingestion of foreign objects.

8.1.12 The TMS has sustained limited plumbing damage and still operated accurately. Tests have yet to be made to determine the extent of system degradation as a function of plumbing damage.

8.1.13 Pressure probes and plumbing on the J85-CAN-15 engine can be leak tested with the engine installed in a CF-5D aircraft by using tools similar to those developed for this project.

8.1.14 Airborne data recordings were invaluable as a diagnostic aid during the flight trials. Computer print-outs gave sufficient data to permit an analysis of the operation of the various components of the TMS. Data plots were used to indicate the relationships between thrust and engine and aircraft variants. A set of data plots is included in Volume II .

8.1.15 TMS operational potential has been examined by applying flight test data to demonstrate a flight test method of determining certain drag coefficients. It was also shown that the TMS is a good indicator of engine performance as a pre-takeoff check instrument. It has a role to play in optimum flight path predictions and optimum manoeuvres. The TMS gives a direct measurement and indication of thrust and a pilot could be directed to fly a profile with thrust or flight path as a control variable. Thrust data could be input to crash recorders and maintenance log recorders.

8.2 RECOMMENDATIONS

8.2.1 A refined and flight tested TMS based upon the concept proven and reported herein is recommended both as a cockpit indicator of engine performance and as a maintenance tool.

8.2.2 A miniature computer should be designed such that the TMS could be installed in a high performance fighter type aircraft and used as a cockpit instrument.

8.2.3 Pressure probe design must be improved when considering a fleet retrofit.

8.2.4 The indicator face markings should be configured according to flight safety and human engineering recommendations.

8.2.5 Data obtained from the TMS flight trials should be input to the Rutowski profile program in order to compare the resulting newly computed profiles with those attained in the flight tests. This would probably aid in proving the advantages of a TMS in this role.

8.2.6 An investigation should be made into the fact that the percent reference thrust indicator does not always indicate 100% when the engine is operated at MIL power. The investigation should include an examination of data acquired from a fleet audit of engine-to-engine performance variations.

8.2.7 Further development of the system should be undertaken such that the TMS can be applied to afterburning turbofan engines.

8.2.8 A bare engine test should be performed to prove the fail-safe characteristics of the TMS. This test would include an assessment of battle damage and plumbing failure effects on the TMS performance.

8.2.9 Bare engine testing should be conducted to test the TMS response to artificially induced engine malfunctions. This test would extend the range of proven usefulness of the TMS.

APPENDIX I

THE COMDEV GROSS THRUST CALCULATION METHOD

1. THEORY

1.1 The gross thrust calculation technique developed by ComDev is applicable to afterburning turbojet engines with continuously variable exhaust nozzles. The following data can be predicted for a specified engine tailpipe/exhaust nozzle type:

> The engine aerodynamic gross thrust for a complete expansion convergent-divergent nozzle.

1.2 The aerodynamic gross thrust of a jet engine is conventionally defined as the axial momentum of the nozzle exhaust gases plus any pressure force which exists due to incomplete expansion of the exhaust gases. This definition, written in equation form, is:

$$F_G = \dot{m}V_8 + A_8\,(P_{S8} - P_{S0}) \qquad \ldots \text{(I-1)}$$

where:

- $\dot{m}$ is the gas mass flow at station 8
- V_8 is the gas velocity at station 8
- A_8 is the nozzle area at station 8
- P_{S8} is the gas static pressure at station 8
- P_{S0} is the ambient air static pressure

1.3 The pressures which are measured in order to determine the engine gross thrust are identified from a modification of the conventional gross thrust Equation I-1. The alternate expression for the engine aerodynamic gross thrust is available using the continuity equation, equation of state and isentropic flow functions:

$$F_G = P_{S0}\; h(\alpha)\; F(\beta)\; A \qquad \ldots \text{(I-2)}$$

where:

- P_{S0} is ambient static pressure
- $h(\alpha)$ is a measure of the engine performance
- $f(\beta)$ describes the exhaust nozzle operation, and
- A is an engine tailpipe area

1.4 The functional relationships in Equation I-2 can be expressed in terms of the dynamic measurements selected as shown in Figure 32, namely:

$$h(\alpha) = f_1 (P_{T5}, P_{S6}, P_{S7}), \text{ and}$$

$$f(\beta) = f_2 (P_{T5}, P_{S6}, P_{S7}, P_{S0})$$

1.5 The term $f(\beta)$ is a function of nozzle pressure ratio, ratio of exhaust gas specific heats, γ, and nozzle configuration. Figure 33 depicts the one-dimensional, theoretical variation of $f(\beta)$ with ratio of specific heat and nozzle pressure ratio for a convergent only and a complete expansion exhaust nozzle.

1.6 The ComDev gross thrust computing system, block-diagrammed in Figure 34 employs engine calibration constants (C_{5-6}, C_{6-7}, E) to assist the tailpipe pressure measurement-based gross thrust determination procedure. The constants are available from ground, static runs of the bare engine.

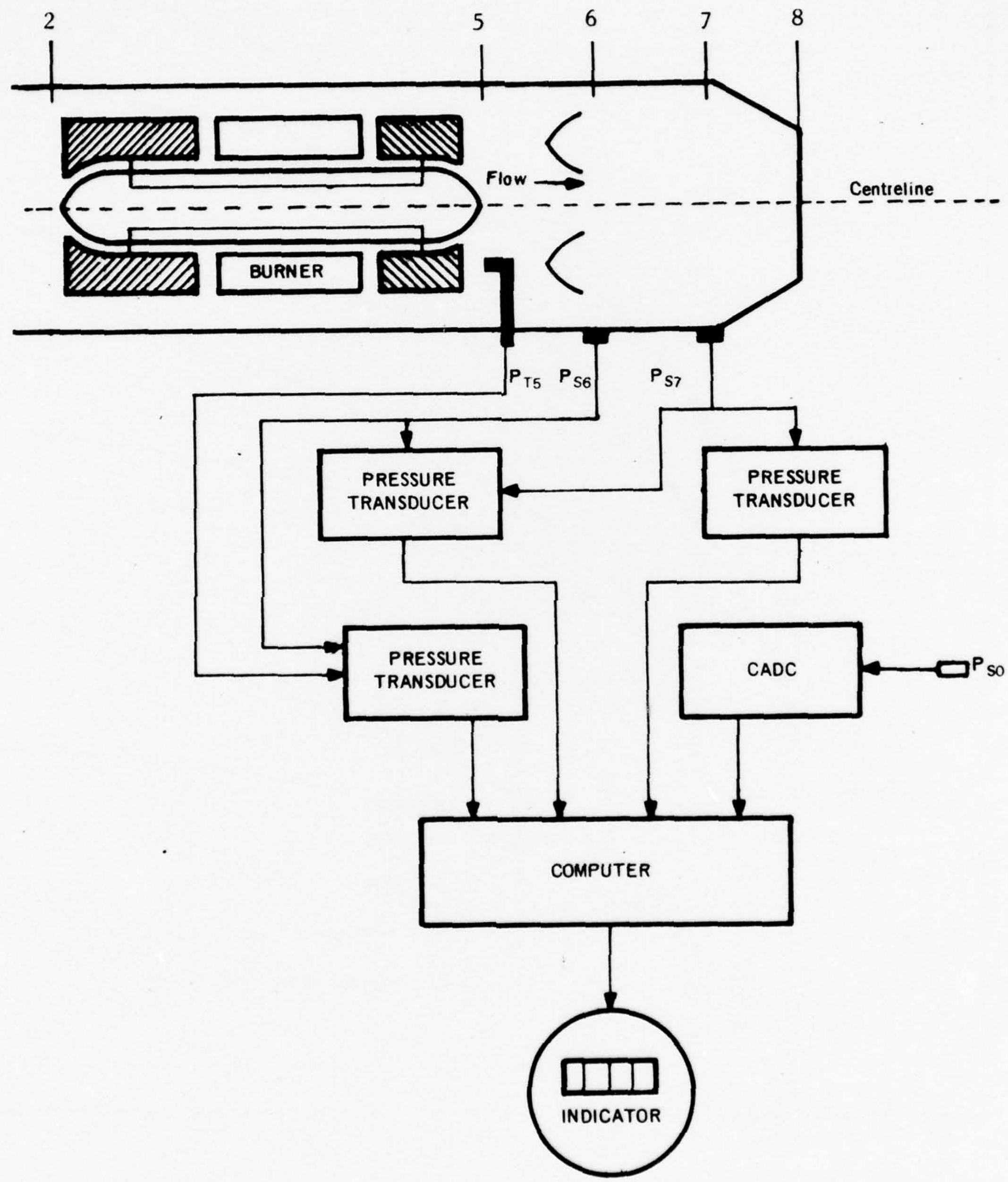

Figure 32: Tailpipe Measurement Station Designation

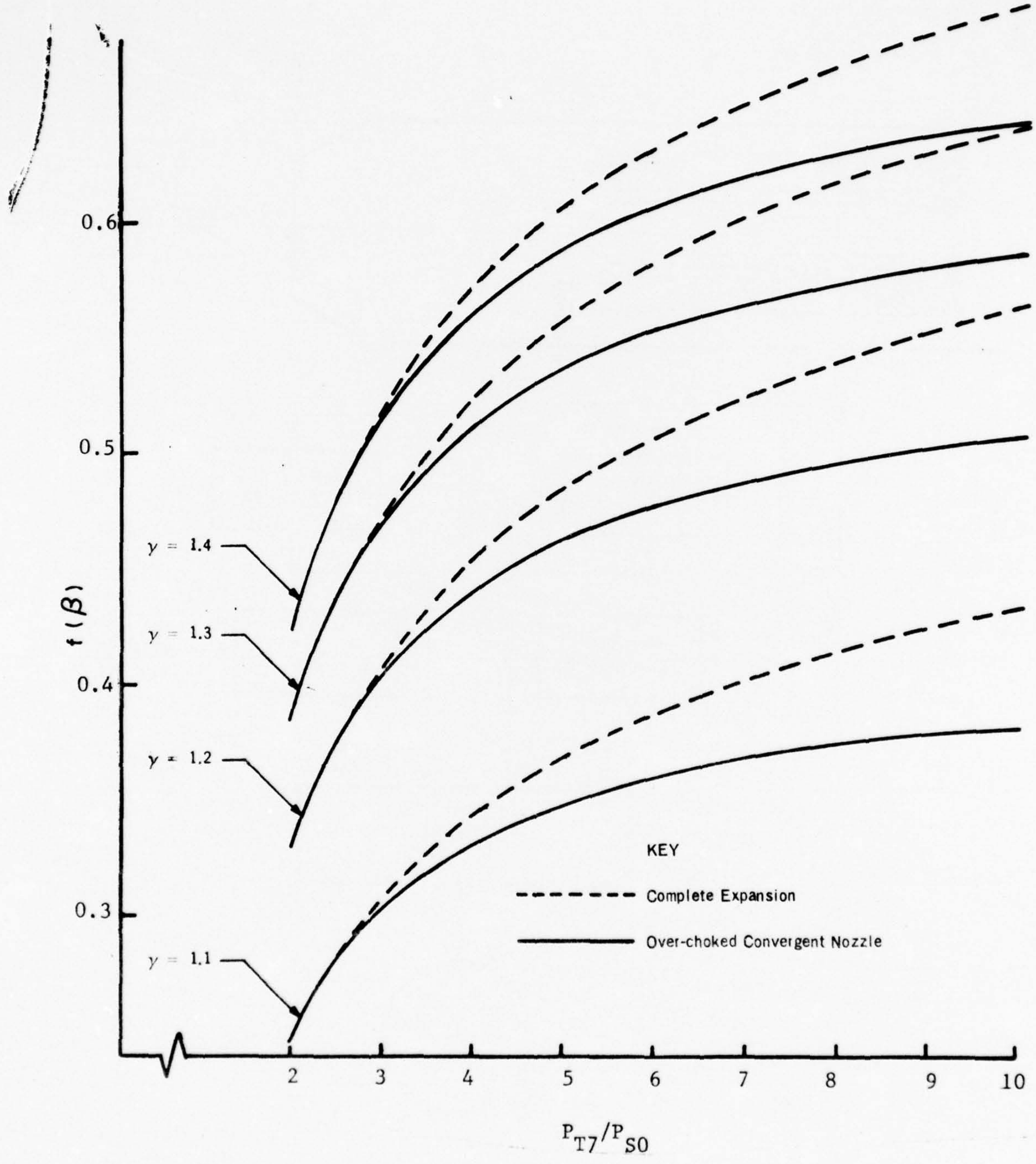

Figure 33: Gross Thrust Parameter vs Nozzle Pressure Ratio for Various Ratios of Specific Heats

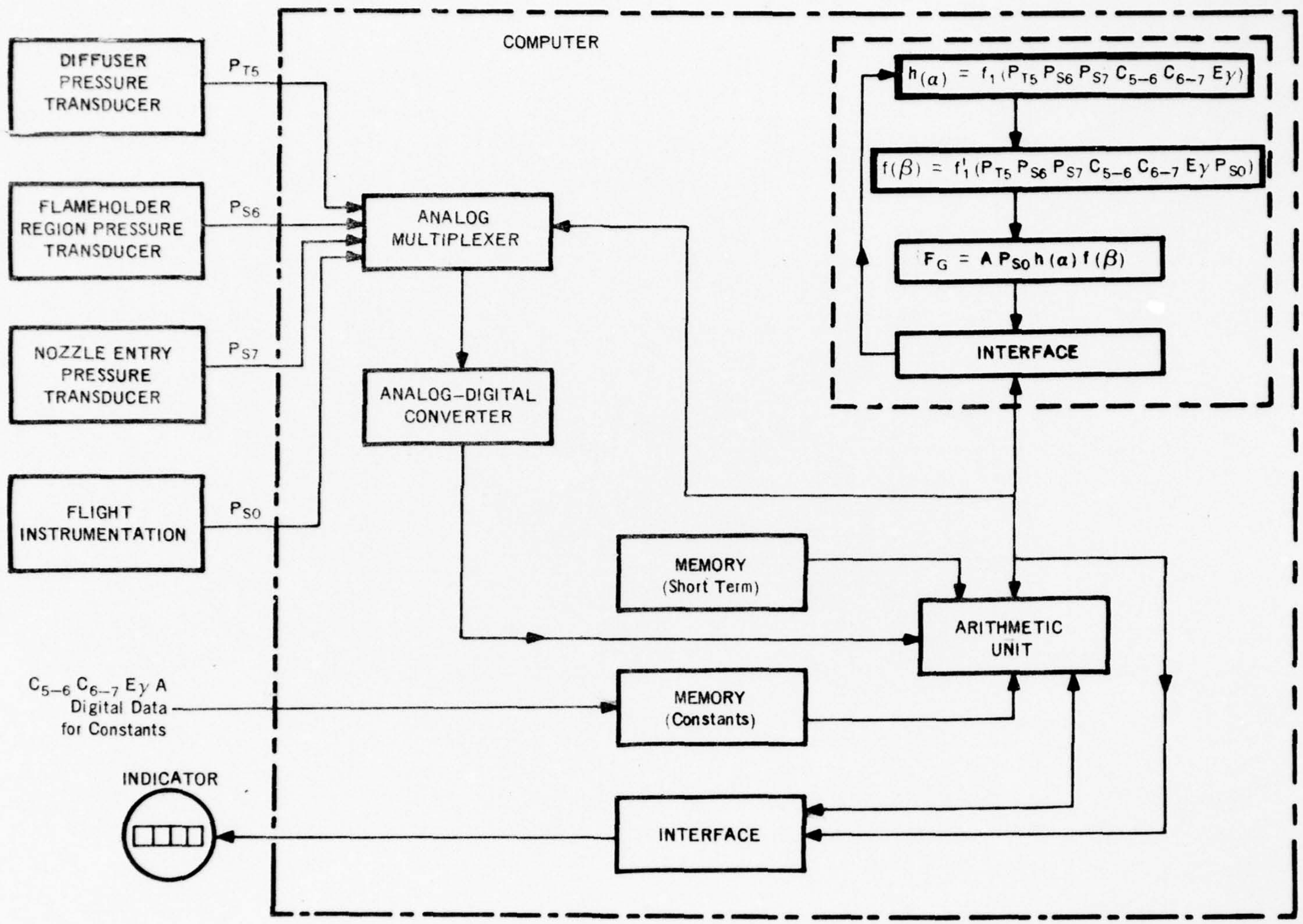

Figure 34: Detailed Gross Thrust Computing System

APPENDIX II

THE COMDEV REFERENCE GROSS THRUST CALCULATION METHOD

1. THEORY

1.1 The reference gross thrust computation provides a value for the primary gross thrust developed at military power by an average engine under the prevailing aircraft flight conditions. The percent of reference thrust indication is therefore the ratio, expressed as a percentage, of the actual engine gross thrust (at the actual power setting) to the available military power thrust (for an average engine) at the same flight conditions.

1.2 For fully expanded flow, the aerodynamic gross thrust of the engine can be expressed as:

$$F_G = \dot{m}_8 V_8$$

Application of the equations of continuity, state and isentropic flow enables the reformulation of this equation in terms of engine parameters and intake recovery. Restricting the analysis to military power operation only, enables these parameters to be defined in terms of aircraft Mach number, altitude pressure and total air temperature, see Figure 35. For the CF-5D airplane at static conditions, the reference gross thrust was computed for the auxiliary intake doors in the open position.

1.3 The required engine and intake recovery parameters were obtained from the manufacturers. An average engine status deck was procured from GE and intake recovery data was obtained from Norair.

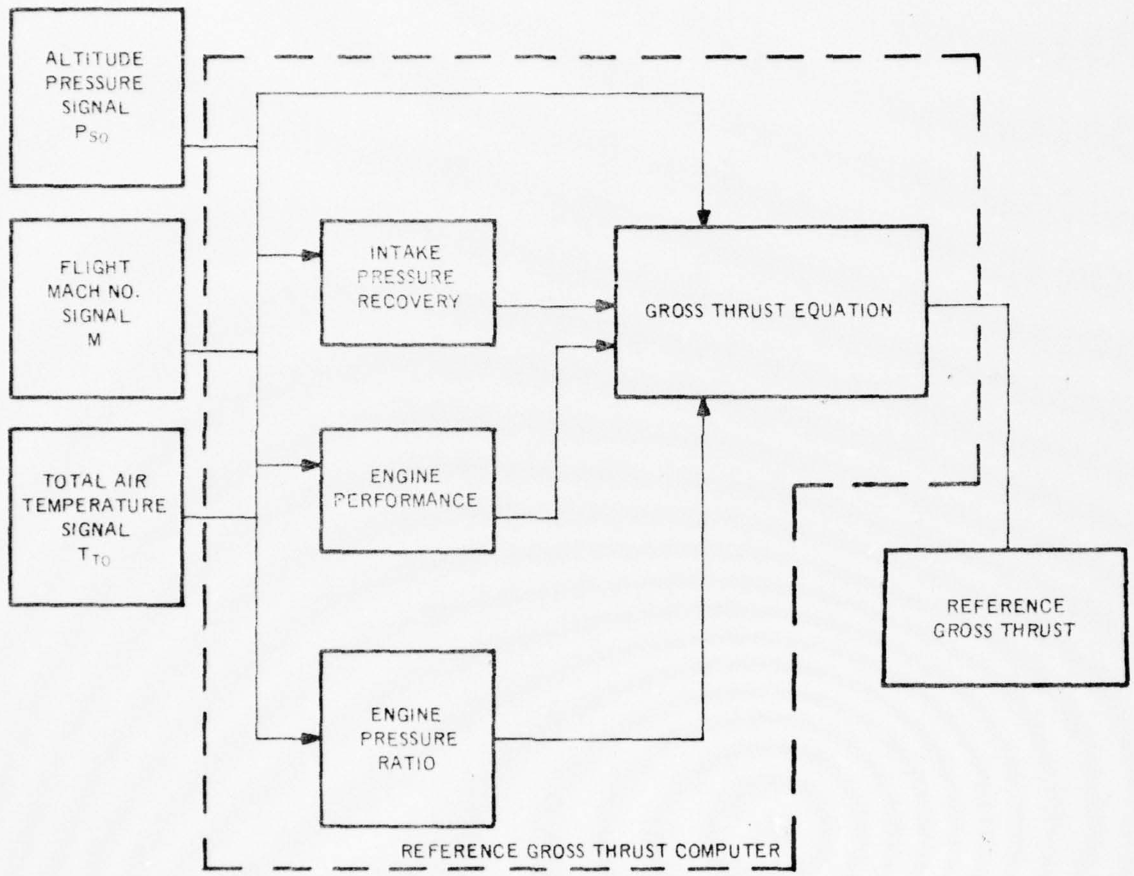

Figure 35. Detailed Reference Gross Thrust System
(Military Power Setting)

APPENDIX III

FLIGHT TEST DATA ACQUISITION SYSTEM

1.1 DATA ACQUISITION SYSTEM

1.1.1 Four distinct data acquisition systems were employed in monitoring the thrust measuring system and aircraft variants. Two systems depended upon observations made by the test pilot, namely pilot observations and taped voice recordings. Two other systems operated independently from each other and, except for on-off actuation, were independent of the pilot. These were a photopanel and a digital tape recording system. These systems are described as follows:

(a) Pilot Observations - CF-5D S/N 116801 aircraft was operated by a QTP who observed aircraft parameters and the thrust indicator during flight trials. He recorded notes on a knee pad test card while conducting a pre-arranged flight mission. Following the mission, he recorded notes at a debriefing.

(b) Ground Recording Station - The pilot maintained a constant radio contact with engineers in a ground station. All radio transmissions were recorded at the ground station. The pilot continuously transmitted a description of his actions and read selected aircraft instruments and thrust indicator readings for recording purposes.

(c) Photopanel - A special panel of instruments located in the rear cockpit contained a thrust indicator in addition to standard engine instruments as listed in Table IX. The pilot actuated a 16mm camera, at selected intervals during the flight, to photograph the panel. A binary clock, on the panel, was synchronized with the digital tape recorder system and provided a means of correlating digitally recorded data and photopanel data.

(d) Digital Recorder - A magnetic tape digital recorder system was located in the rear cockpit. Output of the thrust indicator and aircraft variables were continuously monitored by this system as analog data. At discrete time intervals analog data were converted to digital data and recorded on magnetic tape. Recorded data are listed in Table X. These data were recorded during the entire flight mission at a rate of two complete sets of thrust system data samples per second and one set of aircraft variables per three seconds. This system provided a continuous record of the thrust system performance and the aircraft operating conditions.

2.1 MONITORING OF ACQUIRED DATA

2.1.1 The thrust system computer and indicator performances were monitored by means of the digital recorder and a ground-based data processing computer. Figure 36 is a block diagram depicting the ComDev method of monitoring both the data input to and output from the thrust computer and the thrust indicator. Monitoring proceeded by the following steps:

(a) Mach number and ambient analog data were obtained from the central air data computer in the aircraft. Pressure transducers were used to sense engine pressures and to provide analog data sources. Analog data were passed to the thrust computer and to the data recorder. Both the thrust computer and the recorder had A/D converters to convert analog data to useable digital data. Outputs from both A/D converters were recorded on tape. These data were compared mathematically by the data reduction computer. If significant discrepancies between converters were detected, then warning data were printed during the data reduction process.

(b) The thrust computer solved both gross and reference thrust and passed these data to the indicator and to the recorder. During data reduction, the data processing computer used the same input data and equations as the thrust computer and computed thrusts. These independently computed thrusts were compared with those computed by the thrust computer and percentage errors were printed out. Thus, the thrust computer performance was monitored. (It should be noted that the thrust computer is a 16-bit machine and that data were truncated to 12-bits for tape recorder purposes. Therefore, there will be some round-off error included in the computed error between the thrust and data processing computers).

(c) The thrust system indicator was monitored and indicated thrusts are recorded on tape. Errors between indicator input and needle or digital indications were computed during data processing. Thus, it was possible to detect indicator lag or malfunction.

TABLE IX

PHOTO PANEL RECORDER

Item	Variable	Range
1	Altitude, indicated pressure altitude	0 to 85000 ft
2	Indicated Airspeed	50 to 850 kt.
3	Power lever angle (PLA)	0 to 120^{o}
4	Engine speed (RPM)	0 to 110%
5	Exhaust gas temperature-harness (EGT, T_{5H})	0 to 1200^{o}C
6	Nozzle position (NPI)	0 to 100%
7	Gross thrust (F_G)	0 to 9990 lb.
8	Reference thrust (F_R)	0 to 160%
9	Fail flag on thrustmeter	ON-OFF-FAIL
10	Event marker	ON-OFF
11	Binary coded clock	
12	Run identification	0 - 999
13	Engine main fuel flow	0-5000 pph

TABLE X

DIGITAL RECORDER DATA

Item	Variable	Calibration Range	Recorder(1)* Precision
1	Outside air temperature-TM&R(2)* (T_{TO})(3)*	392 to 780°R	0.1°R
2	Pressure altitude (H_P)	0 to 40000 ft.	19.5 ft.
3	Indicated airspeed (V_i)	50 to 850 kts.	0.4 kt.
4	Mach number - TM&R (M)	0.1 to 1.7	0.0004
5	Outside air pressure -TM&R (P_{SO})	1.687 to 15.236 psia	0.003 psia
6	Power lever angle (PLA)	0 to 114 deg.	0.06 deg.
7	Engine speed (RPM)	35 to 110%	0.1%
8	Exhaust gas temperature (EGT,T_{5H})	0 to 1000°C	0.5°C
9	Compressor discharge static pressure (CDP, P_{S3})	0 to 150 psia	0.07psi
10	Ejector throat static pressure (Ps_{ej})	0 to 20 psia	0.01psi
11	Engine fuel flow (W_{fm})	0 to 3500 pph	1.7 pph
12	A/B fuel flow ($W_{fA/B}$)	0 to 7500 pph	3.7 pph

* See note following list of items

TABLE X (Cont'd)

Item	Variable	Calibration Range	Recorder Precision
13	Nozzle position indicator (NPI)	0 to 94%	0.04 %
14	Fuel remaining - left & right(LFR,RFR)	25 to 2100 lb.	1.0 lb.
15	Event marker	0 to 63	-
16	Clock	-	0.1 sec.
17	Differential pressure input to computer - TM&R (ΔP_S)	-2 to 10 psid	0.003 psid
18	Differential pressure input to computer - TM&R (ΔP)	0 to 10 psid	0.002 psid
19	Absolute pressure input to computer - TM&R (P_{S7})	0 to 60 psia	0.02 psia
20	NAE test stand thrust	0 to 5000 lb.	1.2 lb.
21	Gross thrust indicator output	0 to 9990 lb.	10 lb.
22	Thrust system computed gross thrust	0 to 16000 lb.	3.9lb.
23	Reference thrust indicator output	0 to 160%	0.5%
24	Thrust system computed reference thrust	0 to 180%	0.04%
25	Fail flags (2)	ON-OFF-FAIL	-

NOTES: 1. Recorder precision is the least increment of measurement possible due to the recorder bit length. It does not include calibration precision, manufacturer's tolerance, etc..
2. Variables marked "TM&R" are recorded both as converted by the Thrust computer and data recorder A/D converters.
3. Symbols shown in brackets are frequently used to refer to the corresponding variable.

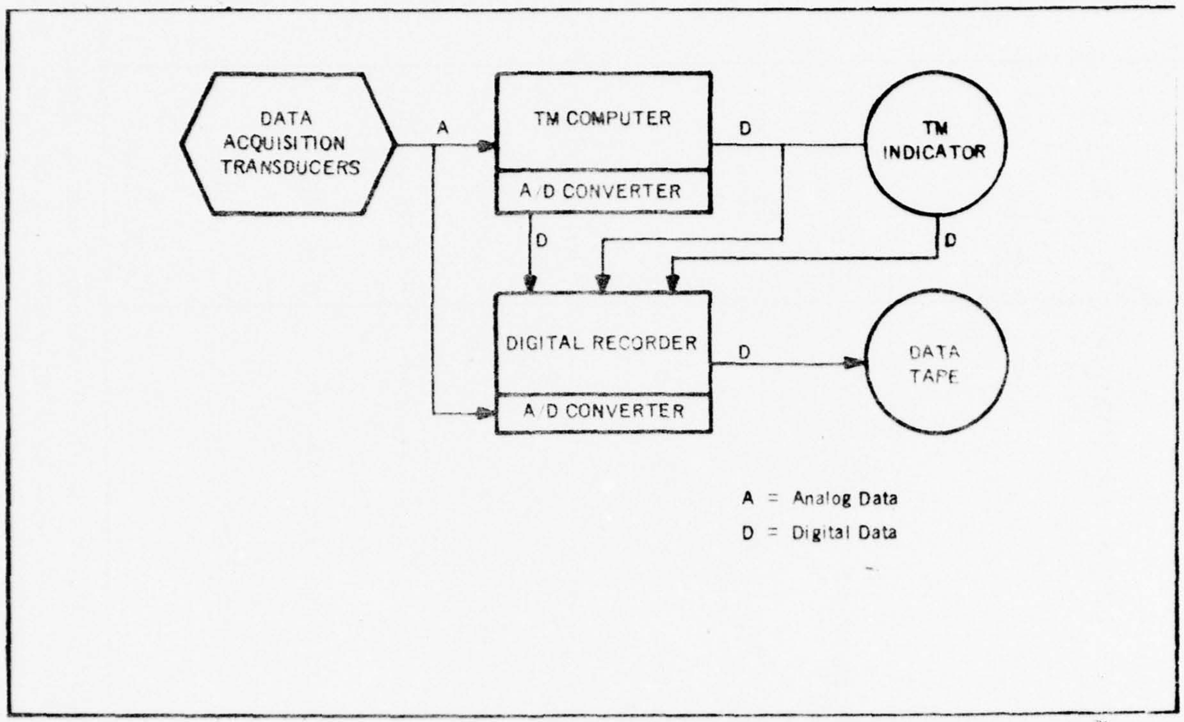

Figure 36: TMS Computer and Instrument Monitoring Method

APPENDIX IV

Appendix IV is a certified true copy of Test Program Project Directive 71/81, Installation and Demonstration of ComDev Thrustmeter. The original document was copied in order to improve the printing quality. Pagination has been modified but the contents of the document are unaltered. Annexes B,C,D and E have been intentionally ommitted.

J.A. Gravelle

J.A. Gravelle

Certified true copy of
Test Program, PD 71/81

R.M. Hallett

R.M. Hallett, Security

RONALD M. HALLETT
A COMMISSIONER FOR TAKING AFFIDAVITS IN AND FOR THE JUDICIAL DISTRICT OF OTTAWA-CARLETON, FOR COMPUTING DEVICES OF CANADA LIMITED, EXPIRES OCTOBER 31, 1973.

APPENDIX IV

TEST PROGRAM PROJECT DIRECTIVE 71/81

INSTALLATION AND DEMONSTRATION OF COMDEV THRUSTMETER

INTRODUCTION

INFORMATION

1. Computing Devices of Canada (COMDEV) is attempting to develop and demonstrate a Thrust Measuring System(TMS) for Gas Turbine engines under a contract with the USAF and the Department of Industry Trade and Commerce (DOITC). The Canadian Forces have loaned a J85-CAN-15 engine to COMDEV as a development tool and the National Research Council is providing technical and test cell support.

2. As a result of initial success in demonstrating the feasibility of such a system, COMDEV has won financial support from the USAF, DOITC and the CF to flight demonstrate the TMS on a CF-5D aircraft.

3. Technical description of the system and development work to date is available in COMDEV monthly Technical Reports held in the Flight Dynamics Section.

4. Further development of the TMS into an engine health monitor or energy maneuverability system could produce an instrument with potential military and commercial applications on future aircraft.

OBJECTIVES

5. The objective of this test program is evaluation of an advanced engineering model of COMDEV TMS in a CF-5D aircraft on a J85-CAN-15 engine in three phases:

a. installation and check-out of the TMS (Phase I);

b. ground static evaluation of the TMS on the NAE aircraft test stand at Uplands (Phase II); and

c. flight evaluation of the TMS throughout the normal operational envelope of the CF-5D aircraft at AETE Cold Lake.

PRIORITY

6. This project is assigned ROUTINE priority.

TARGET DATES

7. Target dates for this program are as follows:

a.	Commencement Phase I (AETE, Cold Lake)	26 June 72
b.	Block leave (AETE)	4 July 72-14 July 72
c.	Commencement of Phase II (NAE, Uplands)	11 Sep 72
d.	Commencement of Phase III (AETE, Cold Lake)	2 Oct 72
e.	End of Phase III (active testing)	27 Oct 72
f.	Completion of Demodification	10 Nov 72
g.	Draft Report	8 Dec 72
h.	Final Report	12 Jan 73

PROJECT AIRCRAFT AND EQUIPMENT

8. CF-5D aircraft S/N 116 801 will be used for the flight trials. This aircraft will be equipped with:

a. an instrumented diffuser cone, after-burner and variable-exhaust nozzle assembly installed on the starboard engine; and

b. the prototype TMS, a digital recorder and a photopanel recorder.

9. All special instrumentation and hardware not immediately available in AETE DA will be provided by COMDEV.

PROJECT PERSONNEL

10. The project personnel assigned to this program are:

a. AETE personnel

(1)	Project Officer	Capt. E. Morin (FD)
(2)	Project Pilot	Maj. G. Smith (AFT)
(3)	Instrumentation Engineer	Capt. B. Turner (AvD)

b. COMDEV test support personnel

(1)	Project Engineer	J. Lafeber
(2)	Data Reduction and Analysis	A. Gravelle D. Dods
(3)	Test Engineer	R. Alexander
(4)	TMS Design Engineers	K.T. Woll T. Rubino
(5)	Electronic Technician	G. Foad

METHOD

SCOPE

11. A test program, including both ground and flight tests will be conducted by AETE to evaluate the COMDEV thrustmeter system. This program is designed to assess the system performance and in particular TMS indicator response to power lever angle, airspeed, and altitude over the flight envelope of CF-5D including ground static conditions.

12. The program will consist of three major test periods:

a. Installation, ground checks, and pre-ferry flight tests to ensure that the TMS, the installed engine "hot end" and the data acquisition systems are operational/serviceable. During that period AETE will be responsible for:

(1) Test Engine Preparation:

(a) installing instrumented "hot end" and associated pressure sensors and manifolds; and

(b) rigging and trimming.

(2) Airframe Modification:

(a) installation of project plumbing (pressure lines);

(b) installation of project wiring;

(c) installation of instruments and equipment support;

(d) installation of modified CADC; and

(e) installation of test engine.

(3) Instrumentation Installation:

(a) installation of TMS;

(b) installation of photopanel recorder; and

(c) installation of data recorder and associated power supplies and signal conditioning units.

(4) System De-bugging and Ground Check-out.

(5) Pre-ferry Flight Tests.

Support personnel from COMDEV will be on site to provide assistance during the a/m activities.

b. Ground static thrust test stand measurements at NAE, to evaluate correlation between thrust indicated by TMS and thrust measured by test stand equipment. COMDEV together with NAE personnel will be responsible for:

(1) calibration of test stand;

(2) installation and alignment of aircraft on test stand (AETE to supervise);

(3) static thrust measurements (AETE to monitor).

AETE personnel will be on site to supervise these activities and provide assistance in de-bugging and operating all systems.

c. Flight trials to evaluate system operation throughout the CF-5D flight envelope. The flight trials will be conducted by AETE and will consist of the following tests:

(1) takeoff;

(2) steady and accelerated level flight;

(3) climb, descent and dive;

(4) maneuvering flight;

(5) power lever transients; and

(6) airstarts.

The test requirements for these trials were agreed upon by AETE and COMDEV during preliminary test program discussion. COMDEV will provide detailed requests for specific test conditions to cover customer and company requirements. The method of conducting these tests and the preparation of the mission profiles is the responsibility of AETE. COMDEV will be responsible for TMS and data recorder maintenance.

13. It is estimated that the following ground and flight test hours will be required to complete this program:

a. Phase I:

uninstalled engine ground runs	3
installed engine ground runs	4
pre-ferry flights	2

b. Phase II:

ferry (OD-OW-OD)	8
static ground runs	6

c. Phase III:

flight evaluation	10

FLIGHT SAFETY IMPLICATIONS

14. In order to satisfy instrumentation requirement for total temperature at turbine exit it is necessary to disable the T-5 Biasing System. Both the front cockpit and photo panel EGT gauges will indicate actual exhaust gas temperature vice biased temperature. The pilot EGT gauge will be modified to show the unbiased "Red Line" temperatures.

15. In event of a system malfunction, there is a possibility of hot exhaust gases leaking into the rear cockpit area. Such an occurrence will excite an alarm system warning the pilot of the danger. He then will select 100% oxygen and shut the test engine down.

TEST SYSTEM AND INSTRUMENTATION

16. The test system will consist of:

    a. TMS system;

    b. Instrumented engine (starboard);

    c. Digital recorder system;

    d. Photo panel recorder.

17. The TMS will compute and indicate gross thrust and percent reference gross thrust using inputs from the engine pressure transducers and from the modified CADC.

18. The digital recorder will record digital signals from the TMS computer as well as analog signals from pressure transducers, CADC, and selected test engine and aircraft conditions for the entire flight.

19. The photopanel will record selected test engine and aircraft instruments at selected flight conditions.

20. TMS system - The thrustmeter system will consist of:

    a. one computer box located on rear cockpit instrument tray;

    b. two thrust indicators located in front cockpit and photopanel;

    c. three engine pressure transducers, plus pressure damping cylinders located in rear cockpit;

    d. three stainless steel engine pressure lines from starboard engine bay to pressure transducers (through dorsal fin);

    e. one modified CADC in forward starboard electronics bay (in lieu of existing unit);

    f. one temperature probe on port side of the aircraft nose (in lieu of existing unit);

    g. one temperature pre-amplifier plus bracket located in nose compartment; and

    h. associated cables with connectors.

21. Instrumented Engine - The instrumented "hot end" to be installed on the starboard engine will consist of:

a. one complete casing, liner and flame holder with 4 static liner pressure taps at afterburner entry (station 6) and exit(station 7);

b. one complete VEN minus actuating cylinders and hydraulic lines;

c. one diffuser cone (with mounting pads) including four strap-on integrating total pressure rakes on the struts (station 5); and

d. external plumbing from pressure taps to junction with airframe plumbing, located on port compressor casing.

22. Digital Recorder System - The digital recorder system will consist of:

a. one Incredata MK II A digital recorder;

b. one 28 VDC power supply; and

c. one COMDEV interface for buffering of digital and analog input signals.

23. Photopanel Recorder - The photopanel will be located in the rear cockpit seat well and will consist of the instruments listed in Annex A.

24. Data will be recorded using the digital recorder and the photopanel. A list of the variables to be recorded is given in Annex A.

TEST PROCEDURES AND SCHEDULE

25. Annexes B, C, and D will outline the procedures and schedule of tests to be carried out during phase I, II and III respectively. These annexes will be available at a later date.

DATA REDUCTION

26. Data reduction requirements for this program will consist of:

a. processing and analysis of the digital recorder data; and

b. processing and analysis of the photopanel films and correlation with the recorder data.

27. Digital recorder data will provide:

a. the main basis for diagnosis of suspected system malfunctions;

b. a complete time based record of selected TMS, engine and aircraft variables; and

c. the basis for correlation with (intermittent) photopanel film records.

28. The photopanel film will provide a record of thrustmeter, and conventional engine condition monitoring instruments versus airspeed, altitude and power lever angle for selected segments of the test flight.

29. Both systems will be operative during the ground static tests and the flight tests.

30. During all phases of the testing COMDEV will be responsible for task (a, para. 26). During phases I and III, the digital tapes will be processed in Edmonton using IBM facilities. COMDEV will arrange for tapes transportation and will have personnel on site to run them through the computer. During phase II, tapes will be processed at Computing Devices own facilities. AETE will be responsible for task (b, para. 26) during both phases I and III. During phase II, COMDEV will be responsible for task (b, para. 26).

31. AETE will monitor all data reduction activities and provide assistance in data analysis.

SUPPORT REQUIREMENTS

AVIONICS DEVELOPMENT SECTION

32. The AvD section will be required to:

a. design and develop the instrumentation required to acquire the data listed in Annex A;

b. provide engineering support to assist FTI technicians in fabricating and installing special instrumentation;

c. ensure TMS and associated instrumentation is compatible with aircraft systems;

d. assist COMDEV in TMS de-bugging and trouble-shooting; and

e. design photopanel face.

FLIGHT TEST INSTRUMENTATION (TSO/FTIO)

33. The FTI section will be required to:

a. install modified CADC and associated wiring;

b. install temperature pre-amplifier and associated wiring, and outside air temperature probe;

c. install front cockpit TMS indicator and pilot's control panel and associated wiring ;

d. instrument photopanel face and assist Project Workshop in installing photopanel in the rear cockpit seat well;

e. repackage rear cockpit mounting tray and install same in test aircraft;

f. Assist Project Workshop in installing stainless steel pressure tubing running from the engine area to the rear cockpit area;

g. maintain test instrumentation throughout the program; and

h. calibrate test instrumentation (indicated in Annex A) before, after, and during the test program as required by the Project Officer and provide the latter with the calibration data.

AIRCRAFT STRUCTURES AND MECHANICS

34. The ASM section will be required to:

a. identify and define a proper routing for the pressure lines running from the engine area to the pressure transducers in the rear cockpit area;

b. design PLA acquisition system (mechanical aspect);

c. design photopanel mounting platform;

d. design mounting arrangement for pressure transducers and dampers; and

e. ensure TMS installation is compatible with aircraft structural integrity.

PROJECT WORKSHOP (TSO/PROJECT WORKSHOP)

35. The project Workshop will be required to:

a. install the a/m pressure lines (assistance provided by FTI);

b. install photopanel and associated power supply in rear cockpit seat well with assistance of FTI personnel;

c. mount engine pressure transducers and damper cylinders in rear cockpit area (as per ASM design);

d. fabricate new photopanel; and

e. assist FTI section in installing test instrumentation as required.

PHOTOGRAPHIC DEVELOPMENT SECTION

36. The Photo D section will be required to:

a. maintain the photopanel camera throughout phases I and III of the program;

b. supply, load, unload and process photopanel films during flight trials at AETE; and

c. supply films during phase II of the program.

AIRCRAFT FLIGHT TEST

37. The AFT section will be required to:

a. provide test pilot for phases I and III flight trials;

b. qualitatively assess the TMS during the flight trials; and

c. ferry test aircraft to Uplands.

AIRCRAFT MAINTENANCE AND SERVICING (TSO/AMO) and (TSO/ASO)

38. The TSO/AMO section will be required to:

a. assemble test engine (c/w COMDEV supplied instrumented "hot end");

b. rig and trim the test engine (including the scheduling of J85 Test Cell run(s) as necessary);

c. install test engine in test aircraft;

d. remove control column (rear cockpit);

e. provide assistance with project instrumentation as required; and

f. provide maintenance and servicing support for the test aircraft throughout the program.

DATA PROCESSING CENTRE

39. The Data Centre will be required to:

a. read and partially analyse photopanel films (16mm) during flight trials at AETE.

ADMINISTRATION

REPORTS

40. A Project Report in 15 copies for limited distribution will be submitted for approval 30 days after completion of active testing. Interim project memorandums for limited AETE internal distribution will be submitted on completion of phase I and II.

COSTS RECOVERY

41. Cost recovery procedures and policy will be as per Annex E to this test program. This Annex is under preparation and will be distributed shortly.

DATA ACQUISITION SYSTEMS

LIST OF VARIABLES

A photo-recorder panel and a digital recorder system will be located in the rear cockpit area to record flight condition, engine operation, and TMS parameters.

The variables recorded by these systems will be as defined below:

PHOTO-RECORDER

Item	Variable	Range	Accuracy
1	Altitude	0-40,000 ft	± 50 ft
2	Airspeed	0-850 knots	± 2 knots
	(Mach)	0-2.0	± .01
3	Power lever angle (test/engine)	0-120°	± .5°
4	Engine speed (test/engine)	0-110% RPM	± .5% RPM
5	Exhaust gas temperature unbiased (test/engine)	0-1200°C	± 5°C
6	Nozzle position (test/engine)	0-100% travel	± 1%
7	Thrust (test/engine)	0-10,000 pounds	± 10 pounds
8	Engine fuel flow (test/engine)	0-5000 pph	± 50 pph
9	Digital clock	0-2 hours	± .5 sec
10	Event marker		
11	Run identification counter		

ANNEX A

DIGITAL RECORDER

Item	Variable	Range	Accuracy
1	Outside air temperature	-60 to 120°C	± 1°C
2	Pressure altitude	0-40,000 feet	± 50 feet
3	Indicated airspeed	0-850 knots	± 2 knots
4	Mach number	0-2.0	±.01
5	Outside air pressure	0-20 psia	±
6	Power lever angle (test/engine)	0-120°	± .5°
7	Engine speed (test/engine)	0-110% RPM	± .5% RPM
8	Exhaust gas temperature at station 5 (T_{t5})	0-1000°C	± 5°C
9	Compressor discharge static pressure (test/engine)	0-150 psia	± .5 psia
10	Turbine exit total pressure (test/engine)	0-100 psia	± .5 psia
11	Afterburner upstream static pressure (test/engine)	0-70 psia	± .5 psia
12	Afterburner downstream static pressure (test/engine)	0-60 psia	± .5 psia
13	Ejector throat static pressure, manifold, (test/engine)	±2.5 psid	
14	Engine fuel flow (test/engine)	0-5000 pph	± 50 pph
15	Afterburner fuel flow (test/engine)	0-10000 pph	± 50 pph
16	Afterburner pilot line fuel flow (test/engine)	ON-OFF indication	
17	Fuel temperature	0-200°F	± 2°F
18	Nozzle position	0-100% travel	± 1%
19	Fuel quantity remaining (L&R)	0-2000 pounds	± 50 pounds
20	Clock		± .5 sec
21	Event Marker		
22	Run identification counter		

APPENDIX V

METHOD OF ACCOUNTING FOR THE EJECTOR FORCE TO OBTAIN THE MEASURED ENGINE GROSS THRUST USING THE NAE GROUND STATIC NET PROPULSIVE THRUST

1.1 THEORETICAL DESCRIPTION

1.1.1 Figure 37 is a schematic representation of the tailpipe of an installed engine typical of a J85-CAN-15/CF-5D configuration. Also depicted in this figure is a portion of the control volume (shown dotted), net propulsive thrust vector (T_M), ejector static pressure (P_{sej}), static pressure at the plane of the nozzle exist (P_{S9}), primary nozzle exit area (A_9), and ejector exit area (A_{ej}). The aircraft environment is assumed to be motionless at the ambient static pressure (P_{SO}).

1.1.2 Writing Newton's Second Law of Motion for this control volume, assuming the cooling air (tailpipe-airframe annulus) axial momentum to be negligible, provides:

$$T_M = (\dot{m}_9 V_9 + A_9(P_{S9} - P_{SO})) - (A_{ej} - A_9)(P_{SO} - P_{sej})$$

where $\dot{m}_9$ = primary nozzle exit mass flow, slug/sec.

V_9 = primary nozzle exit velocity, ft/sec.

1.1.3 The ComDev thrust computing system mechanizes an alternate formulation of the $\dot{m}_9 V_9$ term subject to the assumption $P_{S9} = P_{SO}$.

1.1.4 The ComDev predicted engine gross thrust (primary nozzle only) will differ from the net propulsive thrust (measured at static conditions on the NAE thrust stand) by at least the magnitude of the ejector force. Based upon data shown in the Norair report J85-GE-15 Thrust Calculation, NOR550-15/P15016, Northrop Corporation, Norair Division, the J85-CAN-15/CF-5D static configuration produces $P_{sej} < P_{SO}$, with the resulting necessary condition

F_G (ComDev) $>$ T_M(thrust stand)

1.2 APPLICATION

1.2.1 Application of the ejector force proceeded by the following steps:

(a) Ejector force was plotted vs ambient temperature T_{SO} for military and maximum reheat power settings, with auxiliary inlet doors both open and closed. Data were obtained from the Norair report NOR 550-15 mentioned above. (See Figure 38.

(b) For a given test run the ejector force was obtained at

military and maximum reheat conditions from Figure 38. A linear variation of F_{ej} with T_M was assumed as shown in Figure 39.

(c) The appropriate value of F_{ej} was read from Figure V-3 for a given value of T_M and the equation

$$F_G \text{ (NAE measured)} = T_M - F_{ej}$$

was used to obtain the measured value of gross thrust.

1.2.2 Since the net propulsion thrust measured at military power can vary slightly between test runs, the plot shown in Figure 39 should be re-drawn for each test run. However, in view of the inaccuracy inherent in the data fit, an average line, as shown, was considered satisfactory. For the data gathered with the intake area blockage plate installed, the same plot was used with auxiliary doors open or closed as required. This assumption could be inaccurate but no suitable data were available for this operating configuration. The slope of the curve: maximum reheat, doors closed was estimated since data for only one temperature were available for this condition.

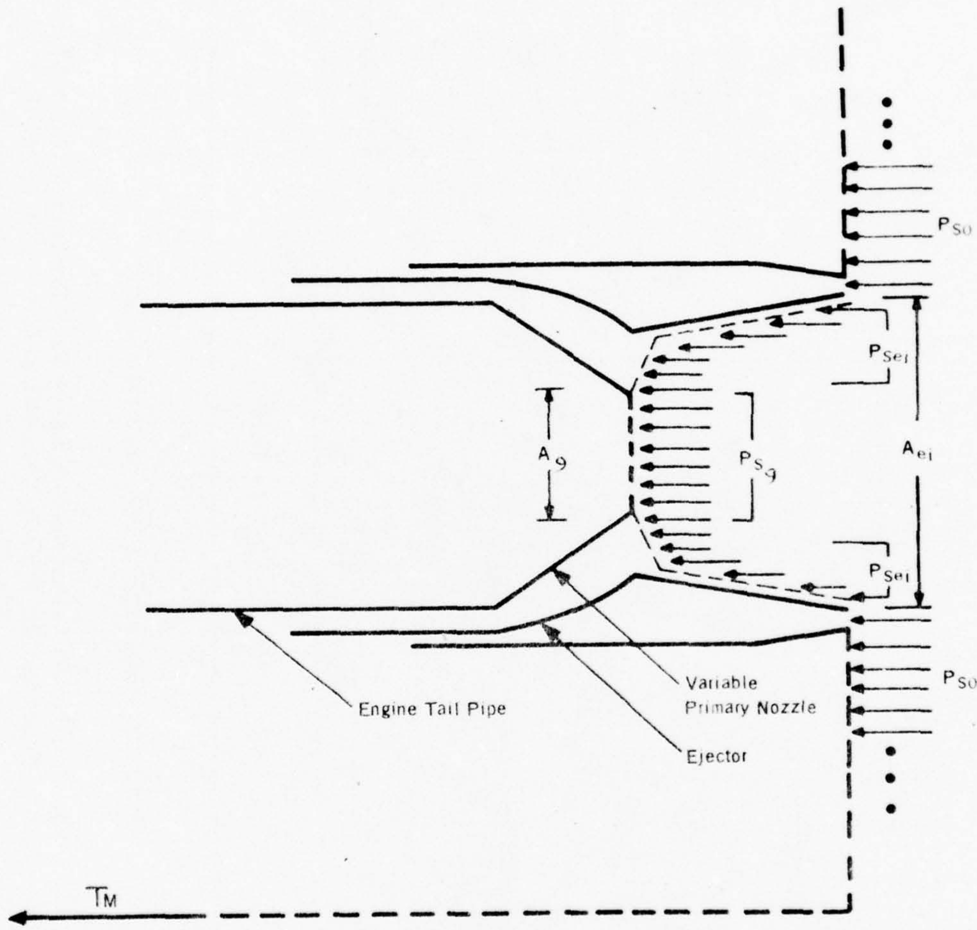

Figure 37: J85-CAN-15 Tailpipe / Nozzle Installed in a CF-5D Aircraft

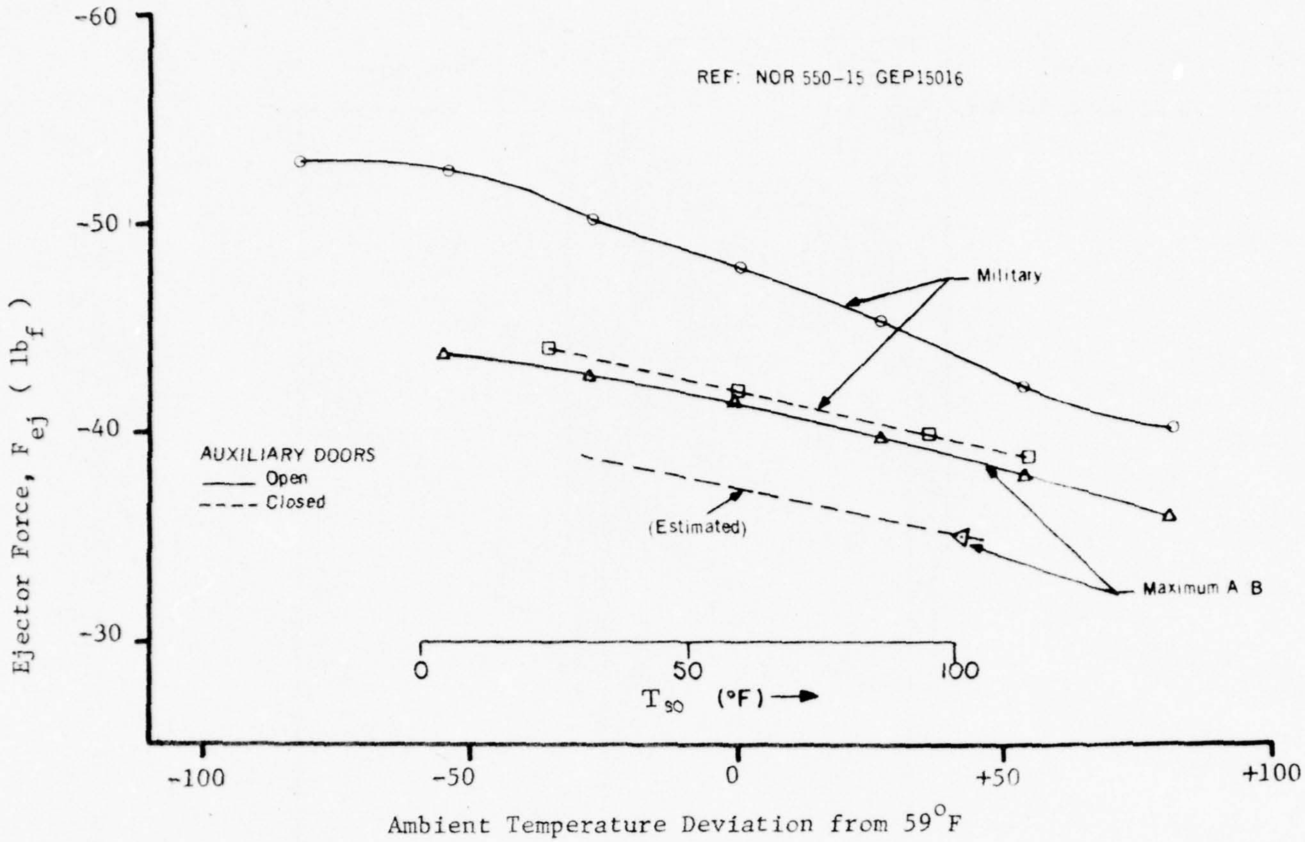

Figure 38: Ejector Force vs Ambient Temperature Deviation from 59°F as a Function of Intake Configuration - Sea Level Static

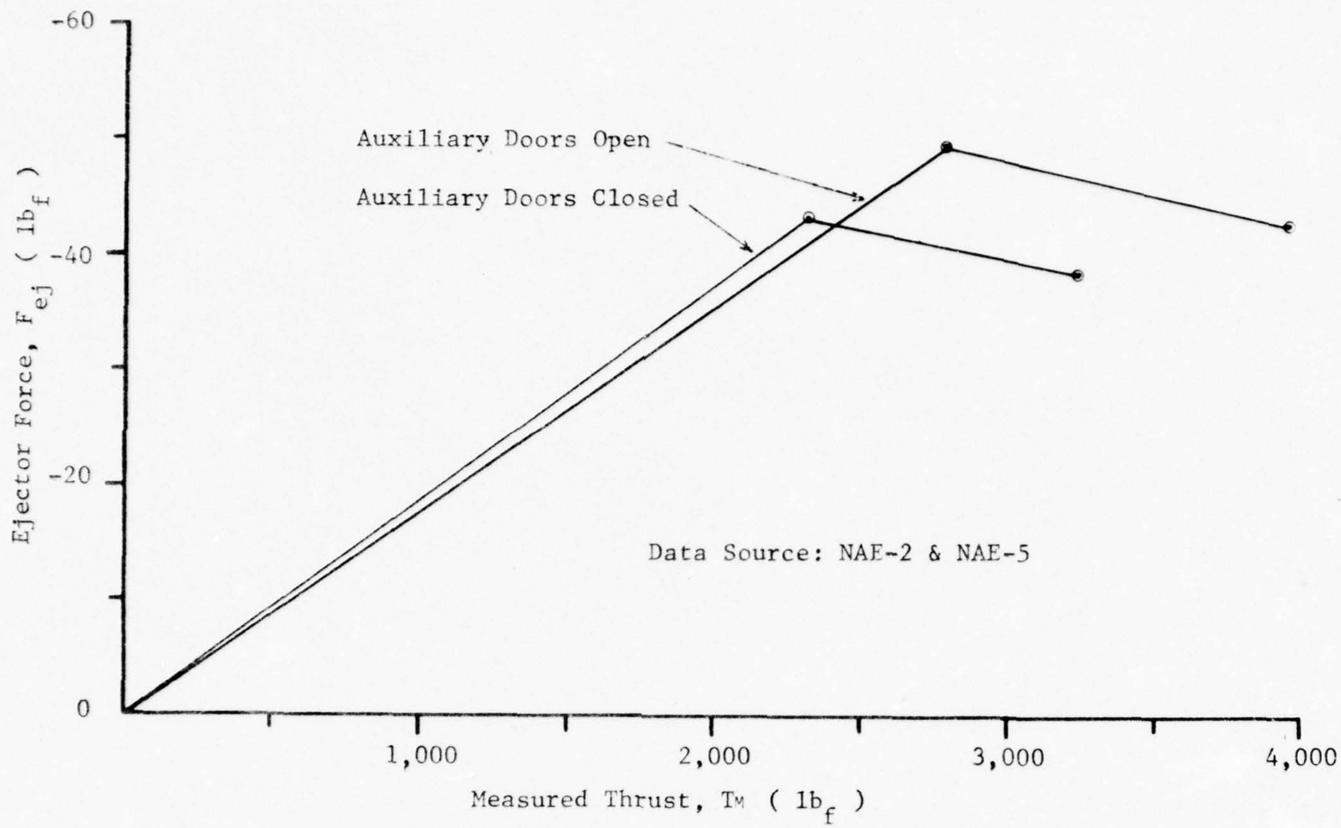

Figure 39: Assumed Ejector Force vs Measured Thrust, J85-CAN-15/CF-5D
Sea Level, Static

APPENDIX VI

J85-CAN-15 ENGINE TRIM

1.1 BACKGROUND

1.1.1 The objective in trimming a turbojet engine is the assurance that the engine will run within specified tolerances at a defined power setting under sea level standard day conditions. The tolerances specified arise from the engine design in order to provide the intended blend of thrust, specific fuel consumption, reliability, maintainability and operating life.

1.1.2 Applied to the Canadian Forces J85-CAN-15 turbojet engine, the trim conditions permit the use of a minimum number of engine cycle parameters to define the desired operating point. Ideally, the engine would be operated at MIL power and under sea level standard day conditions. Since the engine test day differs from the standard day, a table of operating conditions will be required in order to trim the engine under existing conditions. Trimming then becomes a matter of matching the engine operating point to that expected of a nominal engine which had been trimmed under standard conditions but operated under the present conditions. This procedure is implemented by setting the test engine gas generator fuel flow to a target value (±25 lb/hr) as tabulated in Figure 6-1A-27 of EO 10B-10E-2. The engine rotor speed is maintained at 100% ±0.3% mechanical speed for intake total temperatures greater than -14^{o}F. For lower intake temperatures, a corrected speed ($N/\sqrt{\theta_{T2}}$) of 108% is used. (Fuel flow data are tabulated for intake temperatures below -14^{o}F for trim purposes only).

1.2 DISCUSSION OF TRIM PROCEDURE

1.2.1 The engine fuel flow is not adjusted directly. It is caused to vary by changing the limiting value of the indicated turbine exit total temperature T_{5H}, permitted by the engine control system. Changing T_{5H} limit affects the modulation of the variable exhaust nozzle and this influences the fuel flow required to maintain the established rotor speed. The specified rotor speed may also be adjusted. Since the T_{5H} and rotor speed adjustments are to some extent independent, an iterative procedure is necessary to establish the desired engine operation.

1.2.2 The gas flow emerging from the second stage turbine rotor of the J85-CAN-15 engine operating at MIL power will have a radial total temperature distribution which is dependent upon compressor inlet conditions. It is impractical to make sufficient measurements to accurately determine the mean aerodynamic total temperature at the turbine exit, T_{5X}. The J85-CAN-15 engine thermocouple harness temperature, T_{5H}, is obtained by averaging a series of thermocouples at a single radial location. It is assumed that T_{5X} may vary as a function of Mach number and altitude although T_{5H} is constant. Therefore, the temperature ratio T_{5X}/T_{5H} may not be constant with changing compressor inlet conditions. The slope of a linear curve fit to a plot of T_{5X} versus T_{T2} for MIL power and constant T_{5H} defines the engine lapse rate. The lapse rate for the nominal engine will be reflected in the target fuel flow presented for engine trimming.

1.2.3 Figure 40 shows lapse rates derived from the engine status deck P15060 data. Changing compressor inlet total pressure shows no effect on lapse rate. Engine status deck fuel flow data corresponding to these operating conditions are presented in Figures 41 and 42 together with data from EO 10B-10E-2. Both the EO and status deck require the same fuel flow at sea level standard day conditions (see Figure 41) indicating the same aerodynamic mean turbine exit temperatures under these conditions. Since both the status deck and the EO assume engine operation under a constant T_{5H}, the difference between the lines at other compressor inlet total temperatures represents a difference in lapse rates. The lapse rate used to generate EO fuel flow data is not known. There may also be some variance in actual lapse rates between different J85-CAN-15 engines.

1.2.4 Figure 42 shows that as the compressor inlet total pressure is reduced, the EO fuel flow with respect to T_{SO} changes such that it relates more closely to the status deck data. This suggests that the T_{5X} data used to generate EO fuel flow data, unlike the status deck data, are sensitive to compressor inlet total pressure.

1.2.5 Table XI lists the measured T_{5H} temperatures for the project test engine and two other engines which were trimmed to EO 10B-10E-2 specifications. Note that the maximum T_{5H} difference between engines is $48^{\circ}F$ (S/N 8491, S/N 8476) while the variation for the project test engine was $30^{\circ}F$. The test engine lapse rate may differ from the nominal lapse rate provided by the P15060 status deck although the engine was correctly trimmed in accordance with EO 10B-10E-2.

TABLE XI: ENGINE TRIM DATA

ENGINE S/N	TRIM RUN	T_{T2} (°F)	P_{T2} (in Hg.)	T_{5H} (°F)
8491	4 Jul 72	66	28.23	1422
8575	18 Oct 72	38	28.07	1388
8476(test)	16 Feb 72	10	27.07	1380
8476	21 Aug 72	66	28.13	1404
8476	4 Jan 73	-23	27.94	1390
8476	6 Feb 73	0	28.27	1376
8476	26 Feb 73	0	28.24	1374

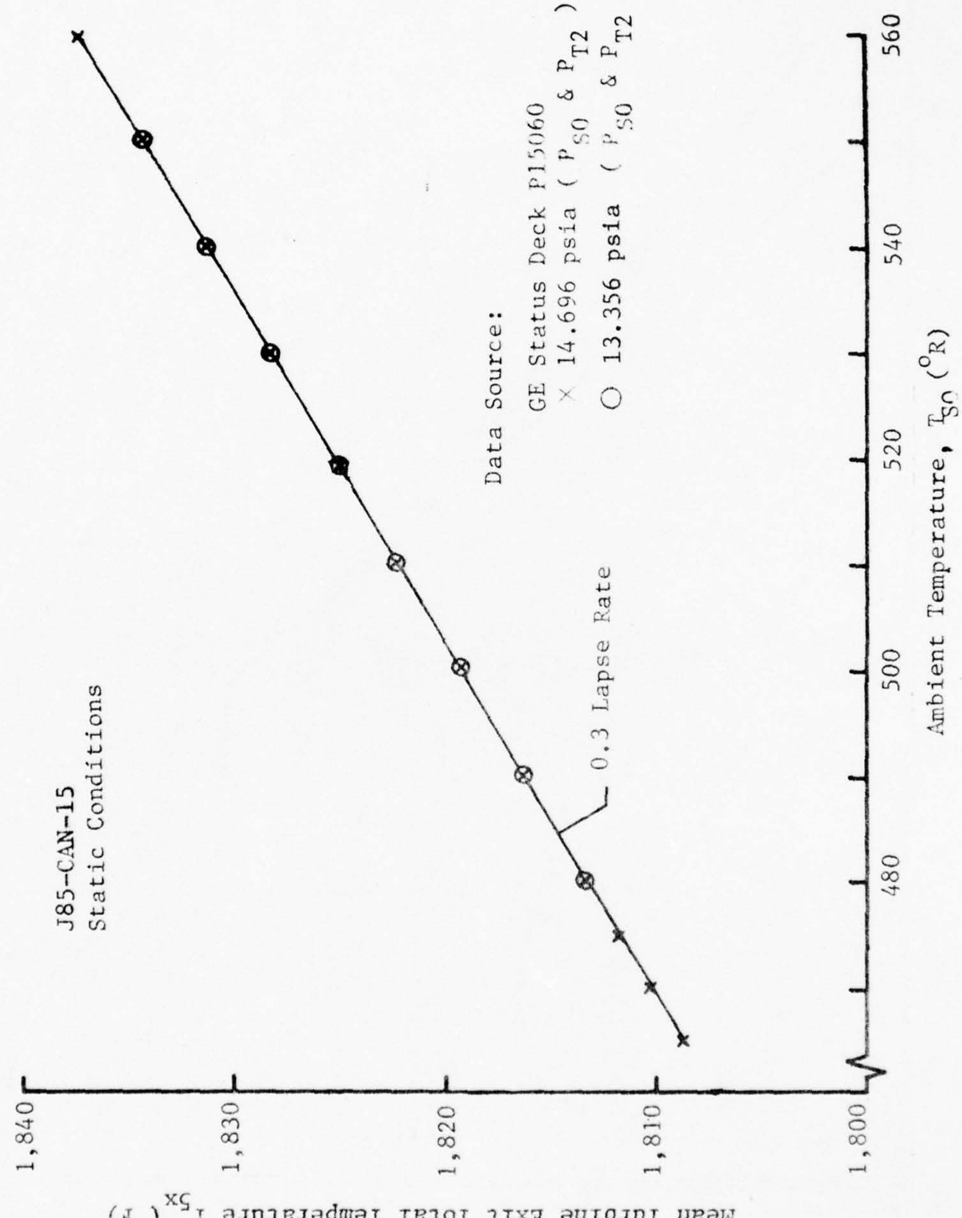

Figure 40: Turbine Exhaust Gas Total Temperature vs Ambient Temperature

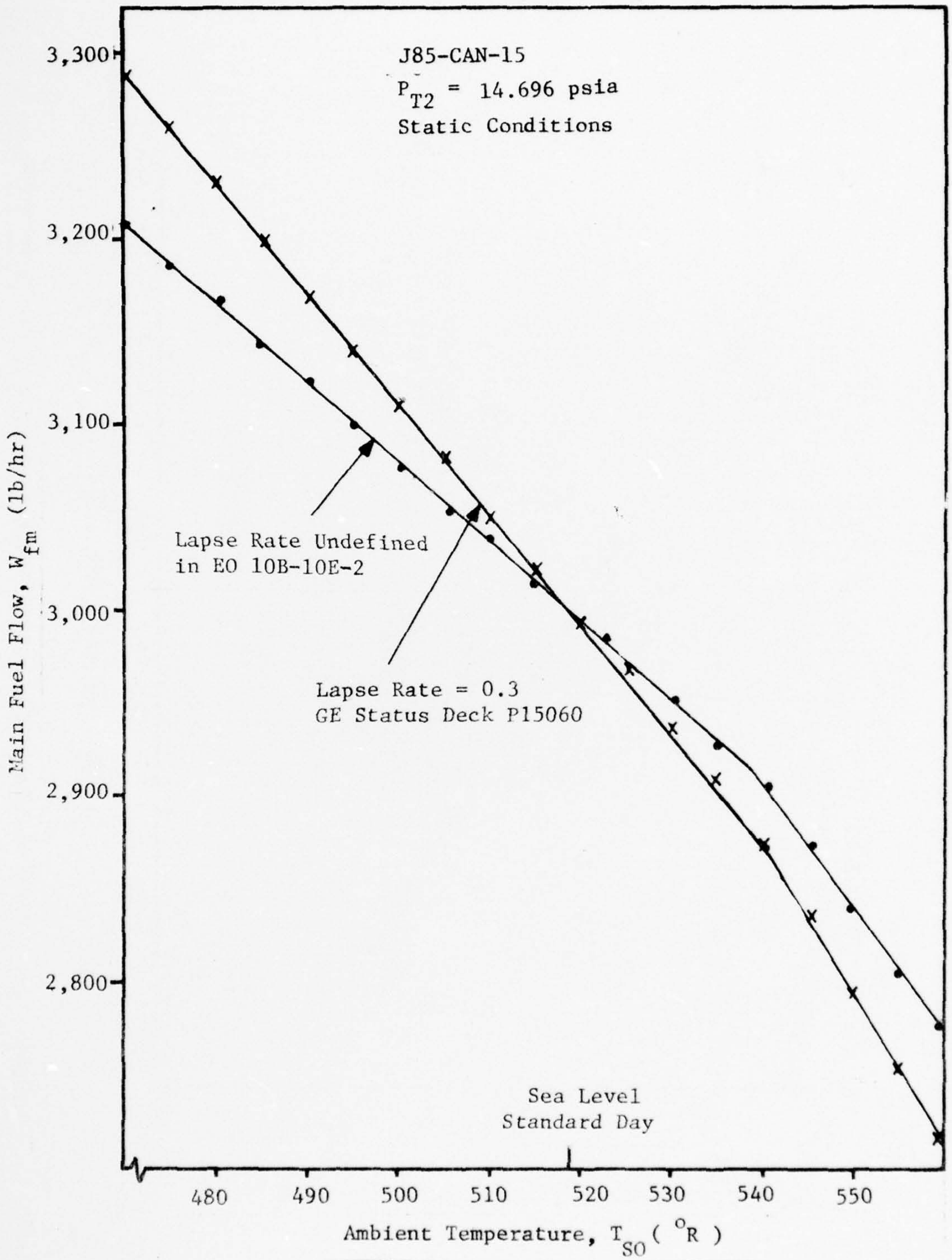

Figure 41: Military Power Fuel Flow vs Ambient Temperature

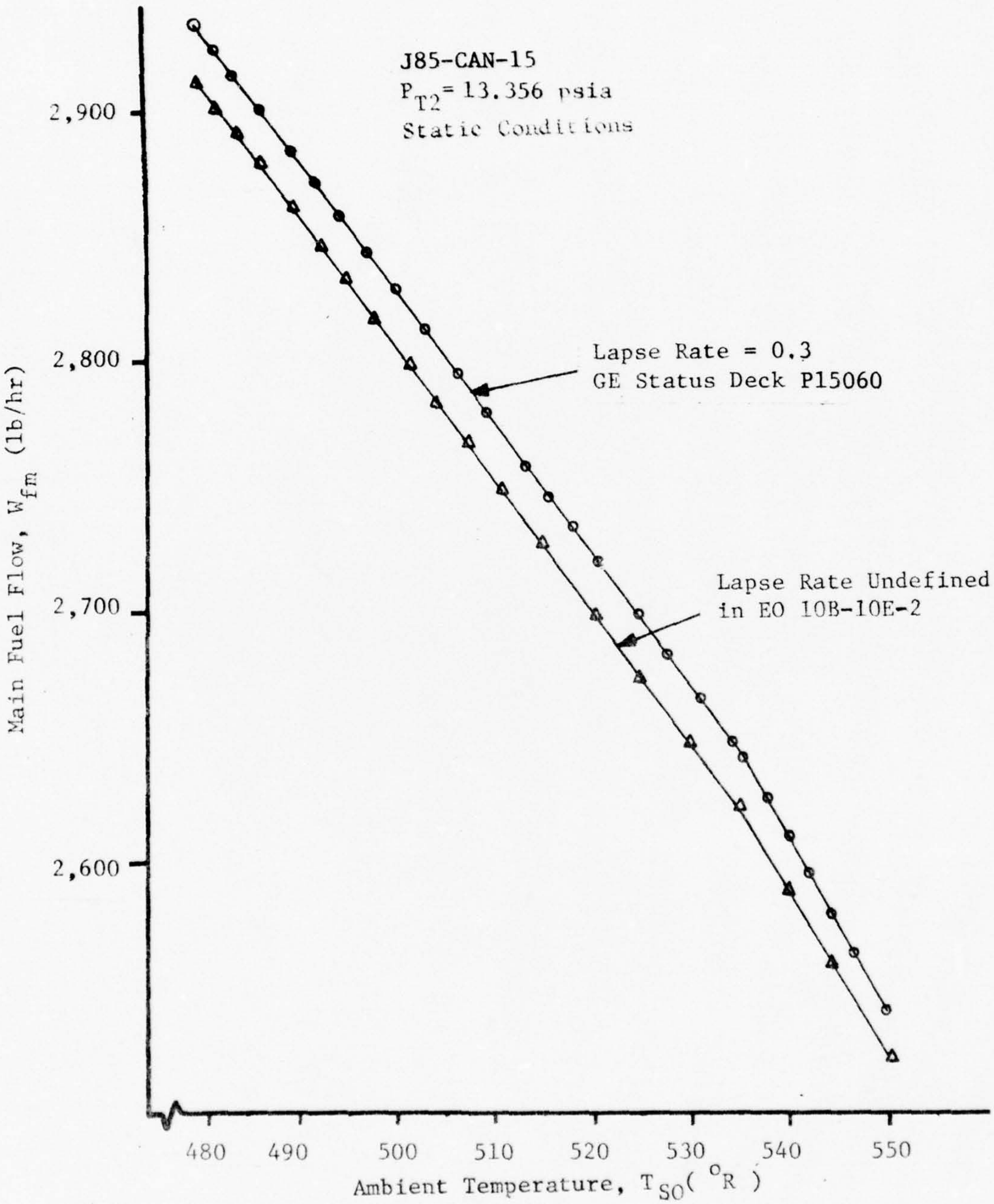

Figure 42: Military Power Fuel Flow vs Ambient Temperature (Reduced Inlet Pressure)

APPENDIX VII

PROJECT TEST ENGINE (S/N 8476) TRIM COMPARED TO
COMPUTER-STORED "NOMINAL" ENGINE (GE P15060) TRIM

1. BACKGROUND

1.1 The ComDev percent of reference gross thrust system operates to display the ratio (as a percentage) of the actual engine gross thrust to the military gross thrust derived from ComDev computer stored engine and airframe intake data. The ratio is computed using the outputs from the CADC and the ComDev measured gross thrust.

2. TRIM CONSIDERATIONS

2.1 The percentage indication using engine S/N 8476 is defined as:

$$\frac{\text{Actual Gross Thrust generated by engine S/N 8476 as trimmed}}{\begin{array}{l}\text{Military Power Gross Thrust at CF-5D S/N 116801} \\ \text{aircraft Mach number and altitude using a "nominal"} \\ \text{or "average" J85-CAN-15 engine (as described by the} \\ \text{G.E. status deck designated P15060 with extracted} \\ \text{horsepower and customer bleed set to zero) receiving} \\ \text{the working fluid at the average total pressure} \\ \text{delivered by an average CF-5D intake duct with} \\ \text{the auxiliary takeoff doors open.}\end{array}} \times 100\%$$

2.2 Engine S/N 8476 is trimmed to the military target fuel flow listed in EO 10B-10E-3. The lapse rate used to generate these fuel flows is unknown. The 'nominal' engine, as specified by GE 15060 performance, reflects the use of a 0.3 lapse rate. It is possible that the reference gross thrust and the actual measured gross thrust could be different at military power because their respective lapse rates could be different.

APPENDIX VIII

CONSISTENCY OF TMS COMPUTED IN-FLIGHT THRUST

1.1 CONSISTENCY OF COMPUTED THRUST

Gross and reference thrust, as computed by the TMS, were recorded during flight trials for data analysis purposes. Therefore, it was possible to use these data in demonstrating the fact that the system was capable of computing consistent data throughout the trial period.

1.1.1 Reference gross thrust is solved by means of a table look-up, CADC data and an equation. Consistent thrust will be resolved provided the computer operates correctly. Computer operation was monitored by recording the CADC variables and the computer output. A ground data processing computer was used to compare the reference gross thrust as computed from the CADC data with that computed by the TMS. Errors between these two computers were printed for each ½ second of the flights. By this means, a number of problems were detected. These problems are described in detail in Appendix IX. Briefly, there were four cases where the reference thrust was computed as one-half the correct value. This problem was believed to be caused by a transient electrical condition. Reference gross thrust was erroneously output as zero on eight occasions for periods of up to one second. A definite cause of this problem has not been established. These problems originate within the TMS computer and are not related to the method of solving reference thrust. With the exception of these cases, reference gross thrust data were consistent throughout the trial. Since a pre-defined equation is being solved, no further consistency test need be conducted.

1.1.2 Gross thrust is solved by the TMS computer from engine data as well as CADC data. These inputs were recorded during the trials and a check similar to that of the reference thrust was made by the data processing computer. Gross thrust anomalies are discussed in detail in Section 6 of Volume II. A number of electronic problems were detected. In the case of gross thrust, it was not considered sufficient to monitor the computer operation in order to show consistency from flight to flight. A TMS should be capable of showing consistent thrust whenever the engine is operated under comparable conditions. An analysis was made of the flight test data in order to demonstrate this consistency.

1.1.3 Level flight performance tests were conducted during several flights. The test pilot set the instrumented right hand engine at MIL or maximum power and used the left hand engine to vary the Mach number. A constant altitude was maintained during the run. Tests were conducted at 5, 20, 30, 36 and 40 thousand feet. Since small deviations in altitude were inevitable, a data analysis had to be made in order to make the data comparable.

These data are plotted in Figure 43. This figure indicates that all the military data points from five flights at various altitudes correlated well with flight Mach number.

1.1.4 A similar analysis was conducted for afterburning trials at altitudes of 20, 30, 34 and 36 thousand feet. Pressure corrections were made as in the MIL power case; the data are plotted in Figure 44. It was noted that most of the data were taken from flights where the power lever angle was 109° to 110°. Flight FT-08 started at a PLA of 110° but was reduced to 108° during the run. Therefore, the faired line was drawn without regard to FT-08 data. A distinct discontinuity occurs in the data in the area of Mach 1.0. This is believed to be caused by a small position error in the Mach number data. Since position error correction data are not available and since a correction would not influence this test, the faired line is acceptable. Flight test FT-02 data were acquired at 34K ft. but were returned to a target altitude of 36K ft. and plotted with FT-08 data.

1.1.5 This consistency analysis compared 200 data points from flights FT-01 to FT-11 over a range of Mach 0.6 to 1.2 and an altitude range of 5 to 40 thousand feet. Since the reduced data seldom deviated by more than one percent of the point from the mean of all the data points, consistency has been shown.

1.2 REFERENCE GROSS THRUST AND GROSS THRUST PRODUCED BY ENGINE S/N 8476

Differences (expressed as a percentage of nominal engine performance) between the computed gross thrust of engine S/N 8476 at military power and the nominal engine gross thrust at military power, ranged between +5% and -6% for pressure altitudes between 1000 feet to 40,000 feet respectively. These differences are believed due to the individual performance characteristics of engine S/N 8476 which could be different from the nominal engine characteristics (i.e. engine performance characteristics predicted by the computer status deck). Generation of the engine status deck is the responsibility of the engine manufacturer.

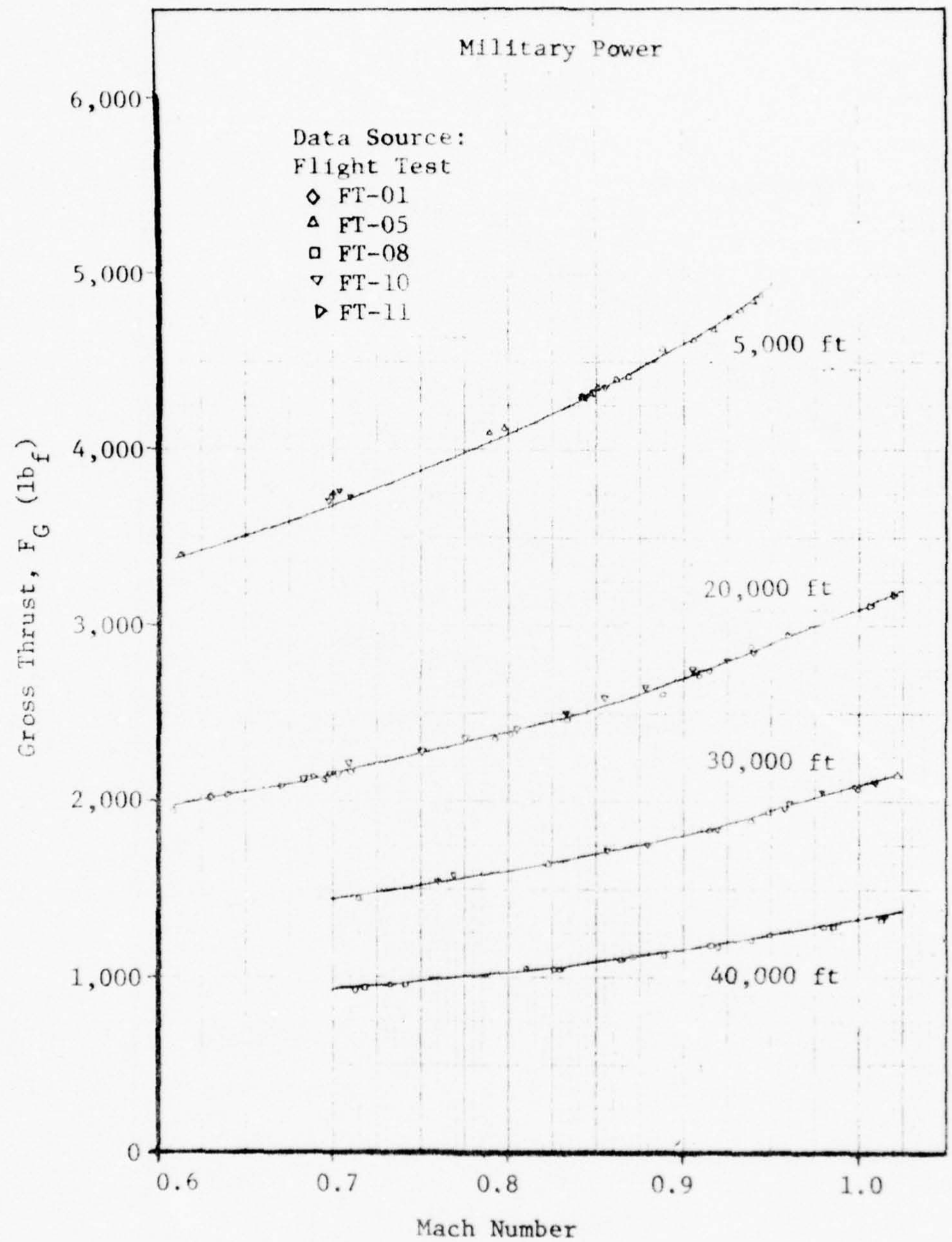

Figure 43: Gross Thrust Reduced to Target Altitude vs Mach Number

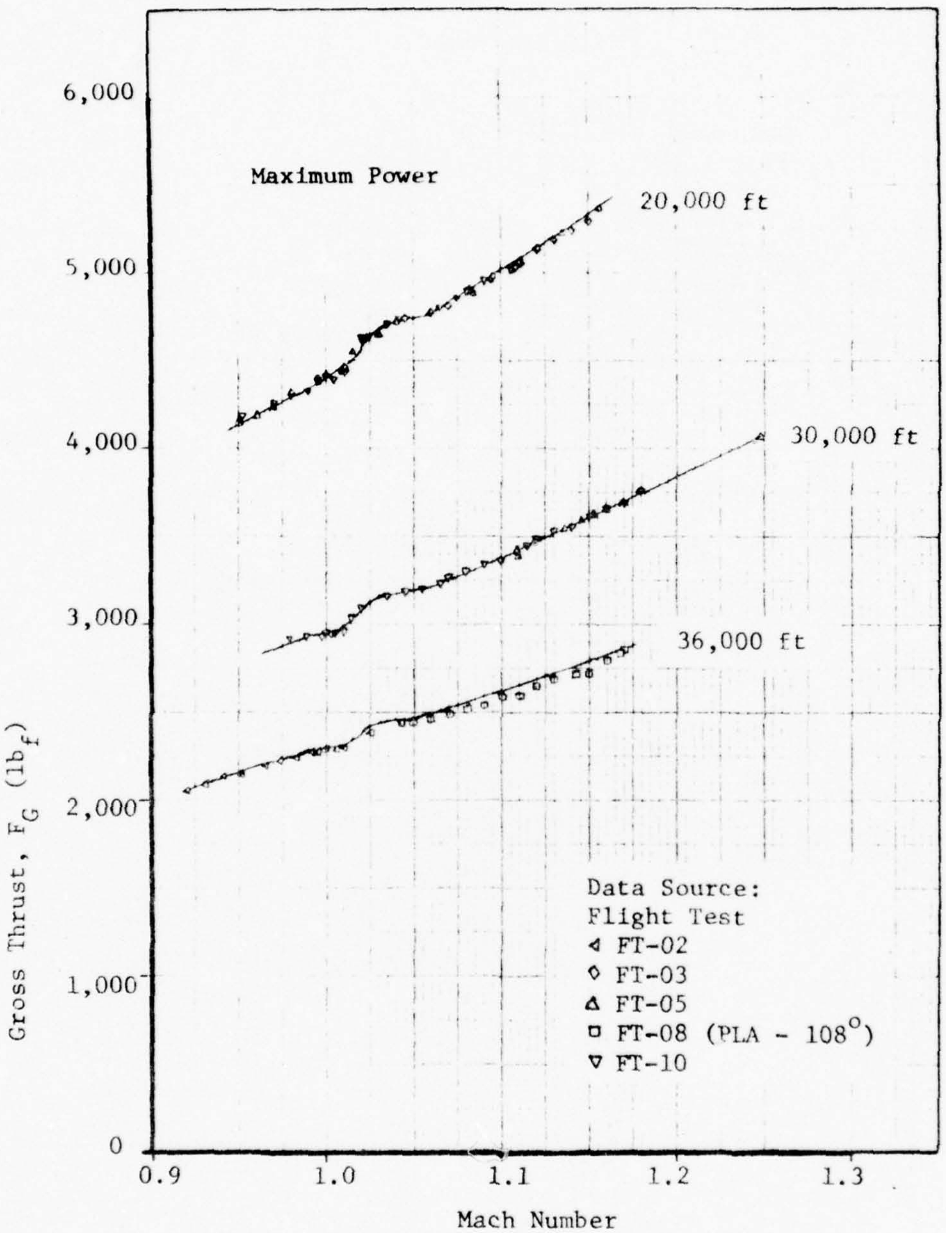

Figure 44: Gross Thrust Reduced to Target Altitude vs Mach Number

APPENDIX IX

ANOMALOUS DATA DIAGNOSTICS

1.1 SUMMARY

1.1.1 Data acquired during the flight trials have displayed a number of problem areas. These problems have been analyzed and found to be related to the following sources:

(a) Recording faults.

(b) Gross thrust computer faults.

(c) Reference thrust computer faults.

(d) Transducer system faults.

(e) Reference thrust indicator lag.

1.2 RECORDER FAULTS

1.2.1 Recording errors may be caused by electrical system transients, G-loading or exceeding the environmental limitations of the recorder. This fault results in the recording of erroneous data for all or part of a ½ second data record. Recording faults do not affect the TMS computer and therefore such records should be ignored.

1.2.2 The recorder formatter was switched off during an air start trial in flight FT-11. This was no doubt caused by an electrical power supply transient. The data reduction computer was forced to process data following this event but plotting was terminated at that point.

1.2.3 Gross thrust exceeded 8000 lb on two flights. The recorder was not designed to record the most significant BCD bit for the most significant digit. Therefore, it was impossible to compute an 8 digit. The plot of indicated gross thrust versus computed gross thrust is in error for these data points. As this is a recording problem, the indicator must not be faulted. A post flight lab test was made in order to prove the capability of the TMS to correctly compute and display thrust in excess of 8000 lb.

1.3 GROSS THRUST COMPUTER FAULTS

1.3.1 The gross thrust computer is inaccurate at thrust levels below 275 lb. The computer is design limited to about 140 to 160 lb gross thrust. Thrust levels below 275 lb cause an overflow condition which results in the computed thrust being erroneously computed as a higher than true value. The computed thrust tends towards the 140 to 160 lb limit as the actual thrust is decreased. This problem becomes even more serious at very low thrust levels where the reference thrust is less than 5%. At

these levels, the overflow condition causes the gross thrust to be periodically computed as an order of magnitude greater than it should be. For example, the gross thrust computation may jump from 170 lb to 1440 lb. This fault was observed some 20 to 30 times during the test flights.

1.3.2 Gross thrust computation errors were observed in the engine transient data. Some 25 errors were observed during power lever angle (PLA) advances and one or two errors were noted during power lever retarding tests. On PLA advance, the computed thrust begins to increase but then drops to a lower value for a period of about ½ a second before increasing again. The computed gross thrust increases on PLA decreasing. It is believed that this problem is caused either by a computer overflow condition or by a delay volume problem. The overflow condition can occur if the computer is used under conditions which exceed its design limitations. The delay volumes were intended to synchronize pressure readings during transient engine operations. These volumes were designed during the engine testing at the NRC engine thrust stand trials. It is possible that the volumes are not entirely satisfactory for the actual aircraft installation. Some further work will have to be done on this problem in future installations.

1.4 REFERENCE THRUST COMPUTER FAULTS

1.4.1 Reference thrusts were erroneously computed as zero on eight occasions. Computed thrust remained at zero for periods of ½ to 1 second. The reference thrust indicator dropped by 47% on one occasion. The indicator may take up to 4 seconds to return to the correct indication following such a fault condition. An investigation has been made into this problem but a definite cause has not been established.

1.4.2 On four occasions the reference thrust was computed as either ½ or double the correct value. This type of error is probably caused by an electrical transient acting like a timing pulse within the computer and this is resulting in a scaling error.

1.4.3 The reference thrust indicator in the photo panel failed during flight test FT-08. It is believed that a computer overflow occurred during engine starting. The overflow caused the indicator to be driven above 160%. An indicator design fault prevents the pointer from returning below 160% once the pointer has passed this point. This problem will be eliminated in future indicators.

1.4.4 One reference thrust computation was 87% but should have been 133%. This problem is believed to be similar to 1.4.2 above.

1.4.5 The reference thrust indicator lags the reference thrust computer due to the mechanical design of the indicator. A special study has been made in order to determine the indicator lag factor. Indicated reference thrust versus computed reference thrust plots show a wide scatter band due to this lag factor.

1.5 TRANSDUCER SYSTEM FAULTS

1.5.1 P_{S7} data fluctuations were observed in the recorded data from flights FT-03 and FT-10. Pressure fluctuations of up to 2 psia were recorded. This caused computed gross thrust variations of up to 400 lb. A sample of one of the periods of P_{S7} fluctuations is shown in Figure 45 Post-flight pressure tests detected a leak in the P_{S6} plumbing. Since both the differential pressures remained constant, it is not probable that the leak was causing this problem. The problem was investigated by plotting P_{S6}-P_{S7} versus P_{S7} pressures in Figure 46. It was observed that P_{S7} data frequently failed to correlate with the majority of the data. These unusual pressures caused the gross thrust fluctuations. It was concluded that either the P_{S7} transducer, the computer circuit or the interconnecting wiring for P_{S7} was failing periodically. These intermittencies were not detected during post-flight static transducer tests. Further bench testing is planned.

1.5.2 P_{T5}-P_{S6} differential pressure data fluctuations were observed in flight test FT-11. P_{T5}-P_{S6} pressures were approximately 3.5 psid at the time but several readings dropped to 2.7 psid and one reading was only 1.3 psid. This caused gross thrust computation fluctuations of 300 to 700 lb. This problem was investigated by plotting P_{T5}-P_{S6} versus P_{S7} data in Figure 47. It was observed that the fluctuating data depart significantly from the mean data. It was concluded that the P_{T5}-P_{S6} pressure transducer, the computer circuit or the interconnecting wiring was probably periodically faulty. These intermittencies were not detected during post-flight static transducer tests. Further bench testing is planned.

1.6 REFERENCE THRUST INDICATOR LAG

1.6.1 The reference thrust indicator is driven by a stepper motor which is actuated by a signal from the TMS computer. The motor starts in low speed and changes to high speed after a brief time interval. An average pointer movement rate of 18% reference thrust per second may be attained by sweeping the needle over its full range. An instrument lag should be expected if the engine thrust change rate exceeds the pointer rate. Data reduced from flight test FT-05, have been used to determine the indicator lag factor, λ, and mean engine thrust rate data.

Data Source: FT-03, 1 hr, 23 min, 0.7 sec

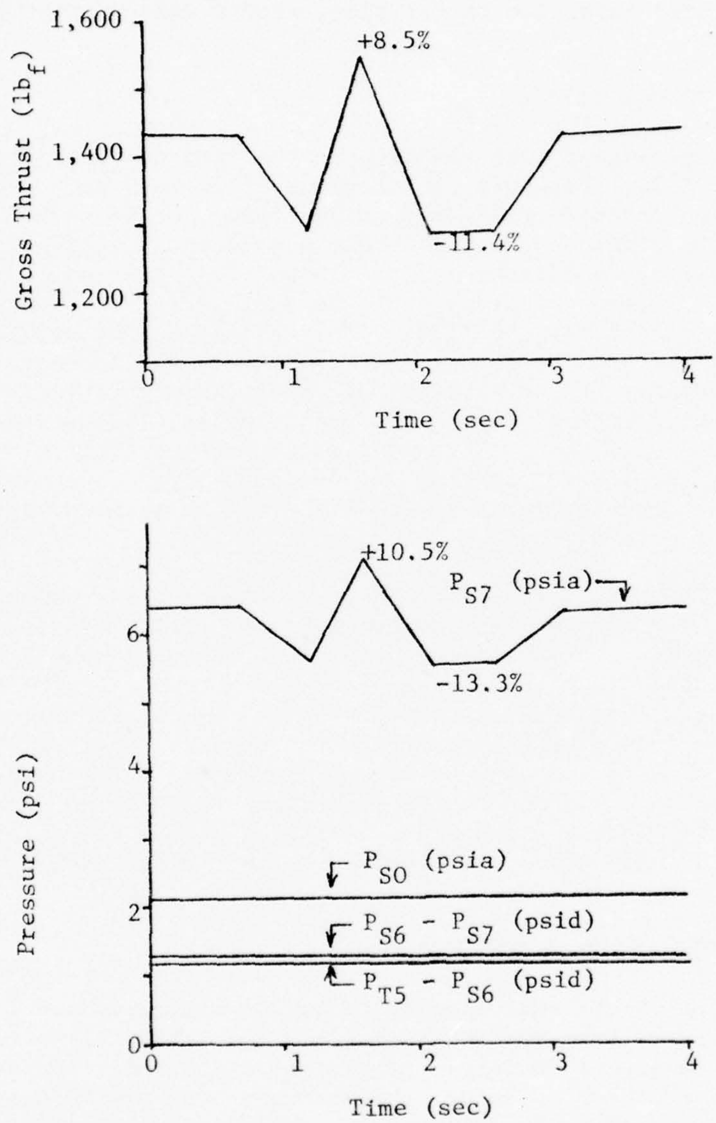

Figure 45: Anomalous Transducer Data - 1

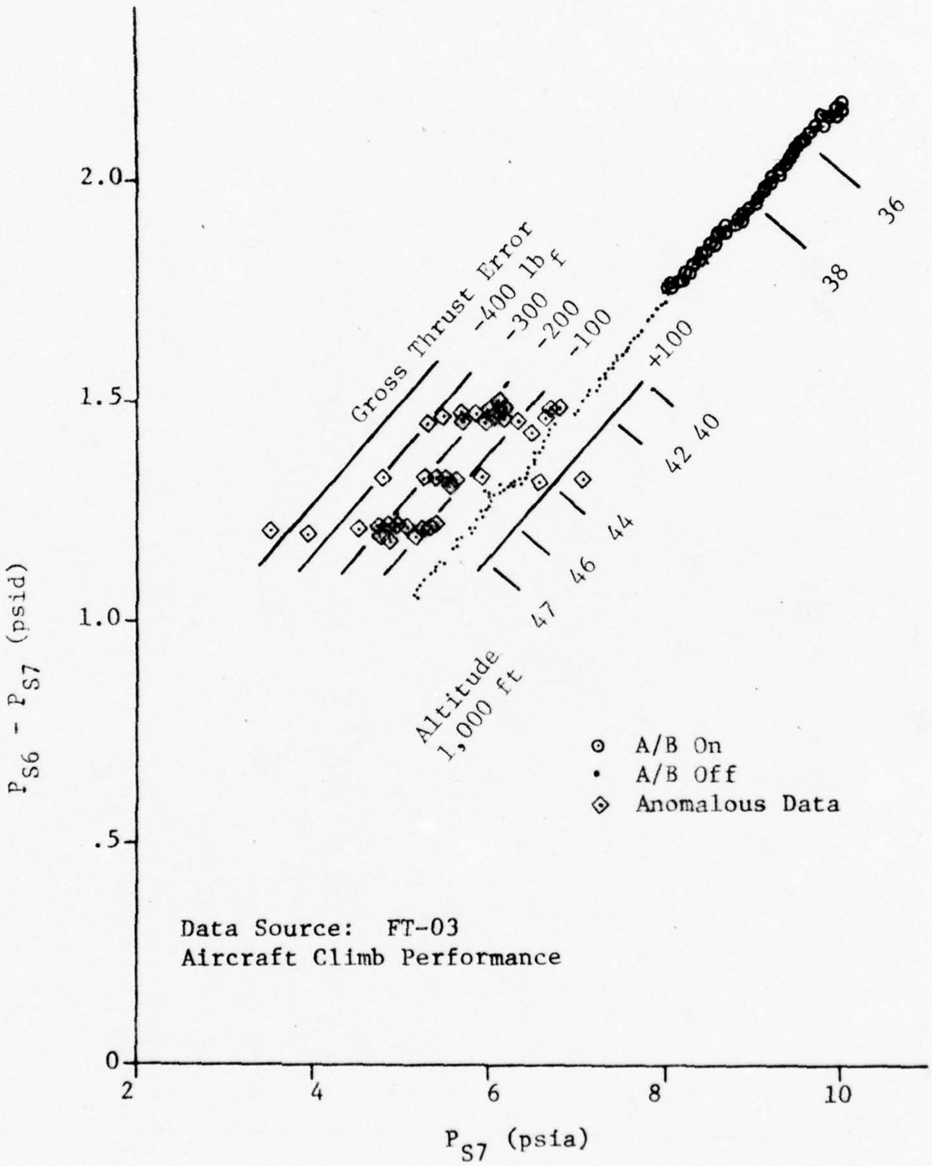

Figure 46: Anomalous Transducer Data - 2

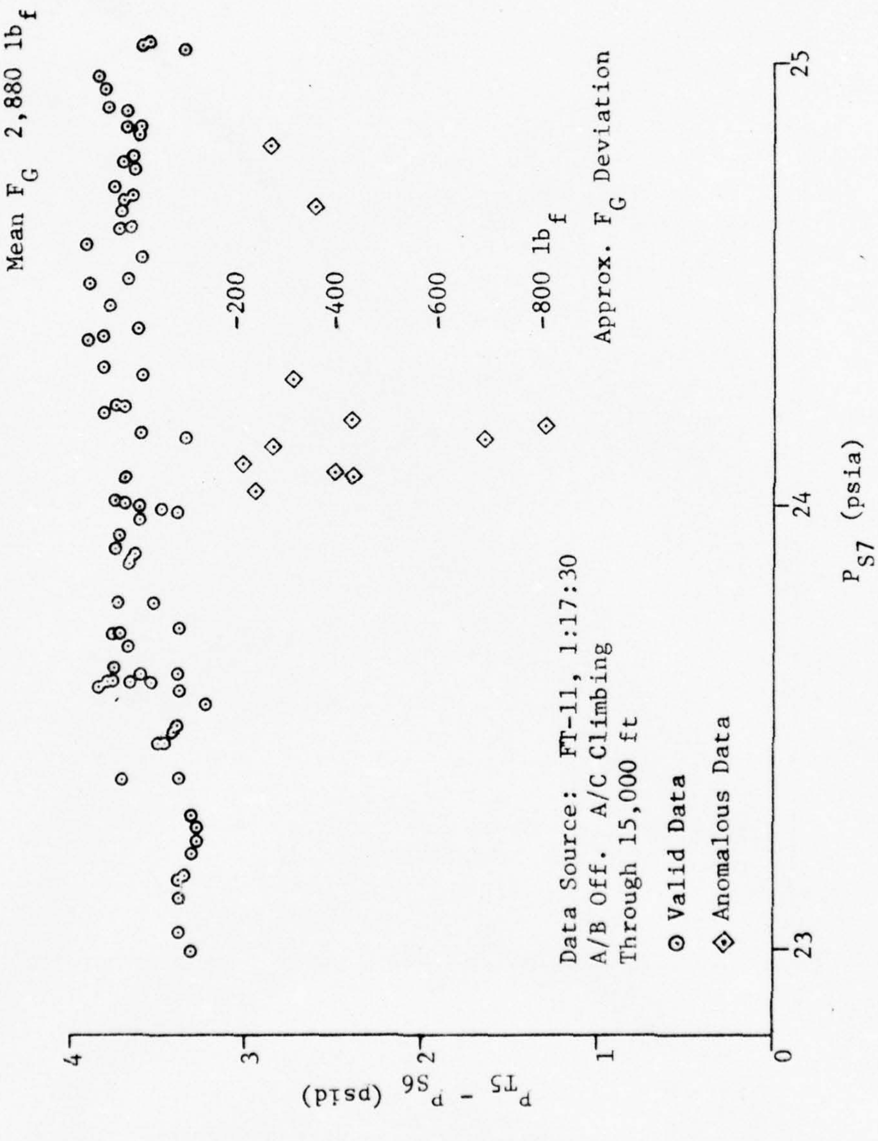

Figure 47: Anomalous Transducer Data - 3

1.6.2 An analog may be made between the TMS reference thrust indicator and an altimeter. The altimeter senses static pressure but is mechanically limited in ability to display altitude correctly during periods of rapid static pressure change. An equation similar to that used for the altimeter may be applied to the reference thrust indicator.

$$\text{Lag factor } \lambda = \frac{(F_G/F_R)_{computed} - (F_G/F_R)_{indicated}}{d(F_G/F_R)_{computed}/dt} \quad \text{sec.}$$

where:

λ = lag factor, sec.

$(F_G/F_R)_{computed}$ = computed percent reference thrust, %

$(F_G/F_R)_{indicated}$ = indicated percent reference thrust, %

$d(F_G/F_R)_{computed}/dt$ = rate of change of computed percent reference thrust, %/sec.

1.6.3 Data recorded during in-flight engine throttle actuations were used to provide plots to solve λ. Figures 48 and 49 were based upon flight test FT-05 data for seven throttle actuations. These data were used to solve the following mean lag factors.

(a) On throttle advancing:

$$\lambda = \frac{10\%}{77/5\ \%/\text{sec}} = 0.65 \text{ sec.}$$

(b) On throttle retarding:

$$\lambda = \frac{-22\%}{-130/4.85\%/\text{sec}} = 0.82 \text{ sec.}$$

1.6.4 It may be seen that on throttle advance, the mean engine thrust change rate is about 15%/sec. Since this rate is less than the mean indicator rate, λ is due to the stepper motor starting period. On throttle retarding, the engine thrust change rate is about 27%/sec. Since this rate exceeds the mean pointer rate, λ will be due to the motor starting and running rate lagging the engine rate.

1.6.5 The mean indicator errors are -10% on throttle advance and +22% on throttle retarding.

1.6.6 Future systems will be built with a much faster reference thrust response. Therefore, the lag factor will not be a problem.

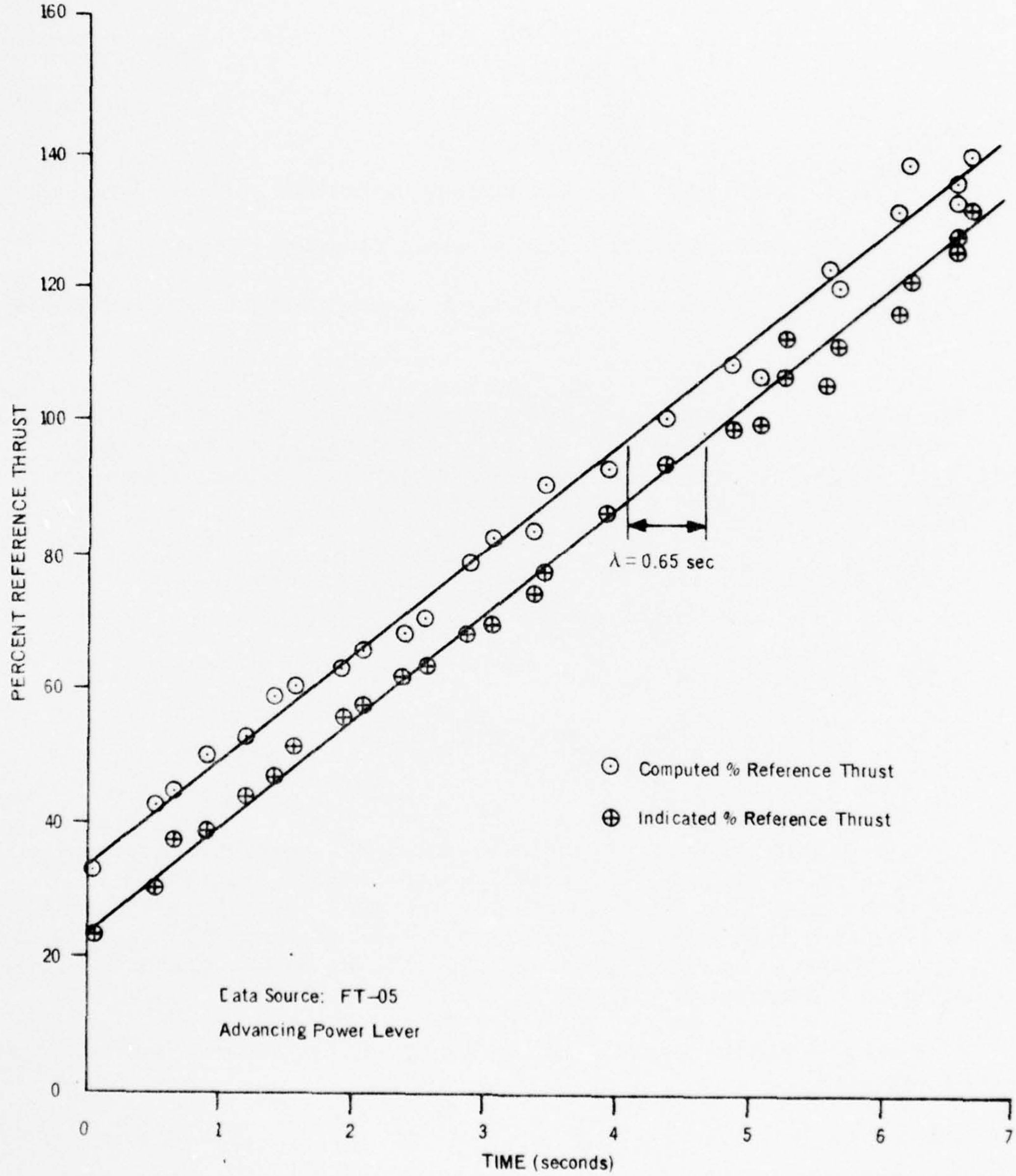

Figure 48: Reference Thrust Indicator Lag

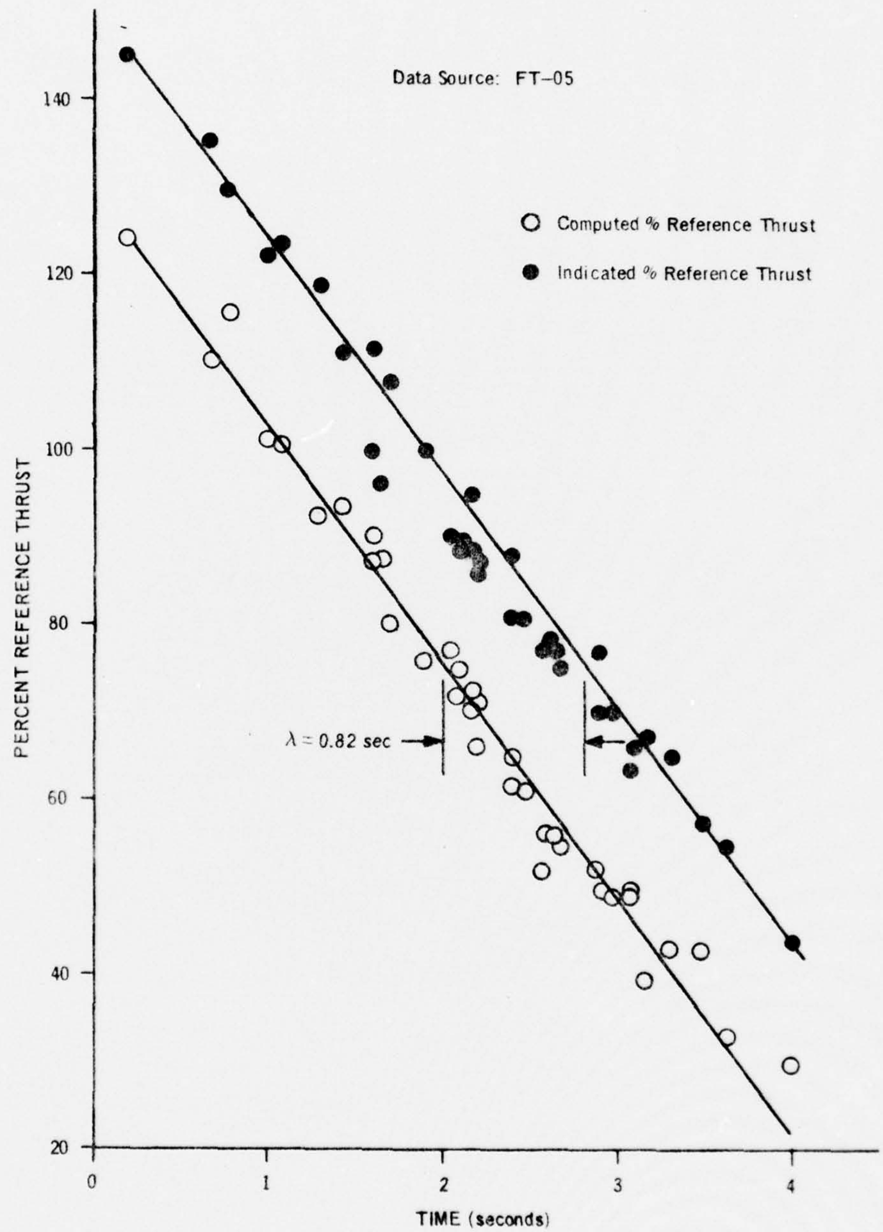

Figure 49: Reference Thrust Indicator Lag - Retarding Power Lever

APPENDIX X

INSTRUMENT FOR DETERMINATION OF DRAG DUE TO AIRPLANE CONFIGURATION ALTERATIONS

1.1 THEORY

1.1.1 Drag due to alterations of an aircraft configuration may be determined with the aid of a thrust measuring system. This could be accomplished by performing a steady state flight in the clean and altered configurations. Any change in net thrust needed to maintain test Mach number will be due to the incremental drag of the configuration alteration. An incremental drag coefficient may be defined as follows:

$$\Delta C_D = \frac{\Delta F_N}{0.7\, M^2\, P_{SO}\, S}$$

where: ΔC_D = incremental drag coefficient

ΔF_N = incremental net thrust, lb.

M = Mach number

P_{SO} = ambient static pressure, lb/ft^2

S = wing area, ft^2

1.1.2 ΔF_N could be approximated by ΔF_{GROSS} if the change in ram drag is not significant. Ram drag may be estimated by means of engine data and engine RPM, ambient temperature, P_{SO} and Mach number.

$$F_{RDG} = \dot{m} V_0$$

where: F_{RDG} = ram drag, lb

$\dot{m}$ = slugs of air/sec at the compressor face

$V_0 = 49.022 \sqrt{T_{SO}}\ M$

T_{SO} = ambient static temperature ^{O}R

1.2 TEST FLIGHT TO DETERMINE ΔC_D DATA

1.2.1 The above theory was tested during flight test FT-09. The test was performed at a selected Mach number and altitude. As only the right hand engine was instrumented the left hand engine was held at 85% RPM during this trial. A reference point was event marked on the data tape at Mach 0.38. Speed brakes and flaps were deployed in various combinations and the Mach number was restored by adjusting the right hand engine. ΔC_D data were computed using the gross thrust data and again using net thrust data. The results of this trial are shown in Table XII.

TABLE XII - COMPUTED ΔC_D DATA						
CONFIGURATION	F_G (lb)	F_{RDG} (lb)	M	P_{SO} (psia)	ΔC_D (1)	ΔC_D (2)
1. CLEAN	1189.0	419.5	0.3829	12.20	-	-
2. SPEED BRAKES	1883.7	478.8	0.3824	12.22	0.022	0.020
3. LE FLAPS	1091.3	405.0	0.3823	12.21	-0.0030	-0.0026
4. FULL FLAPS	1450.0	437.6	0.3832	12.17	0.0084	0.0078
5. LANDING GEAR	2253.7	493.4	0.3830	12.23	0.034	0.032
6. SPEED BRAKES PLUS FULL FLAPS	2078.6	489.4	0.3816	12.20	0.029	0.027
7. SPEED BRAKES + LANDING GEAR	2828.0	488.3	0.3755	12.21	0.056	0.053
8. FULL FLAPS + LANDING GEAR	2473.3	494.3	0.3831	12.21	0.041	0.039

(1) Neglecting F_{RDG}

(2) Including F_{RDG}

1.3 INCREMENTAL DRAG COEFFICIENT COMPUTATIONS

1.3.1 ΔC_D values have been computed for various aircraft configurations by using flight test data. Ram drag effects were demonstrated by computing ΔC_D both with and without F_{RDG} values. ΔC_D data were computed by using the following formula.

$$\Delta C_D = \left(\frac{F_G - F_{RDG}}{0.7M^2 P_{SO}\ S}\right)_{TEST} - \left(\frac{F_G - F_{RDG}}{0.7M^2\ P_{SO}\ S}\right)_{CLEAN}$$

1.3.2 The data of Table XII agree well with each other. For example, ΔC_D due to speed brakes, 0.020, plus ΔC_D due to landing gear, 0.032 , are almost equal to ΔC_D due to speed brakes plus landing gear, 0.053.

1.3.3 Data of Table XII have also been compared with data presented by C.H. Vance in the AOI Performance Substantiation CF-5/NF-5 Tactical Fighter with two J85-CAN-15 engines, NOR-67-30, Northrop Corporation Norair Division, April 1968, unclassified. The aircraft weight was computed to be 11,123 lb at the time the reference clean data were recorded. The lift coefficient for this weight and Mach number of 0.38 at 12.20 psia is 0.355. The following data compares the total drag coefficient computed from the clean aircraft plus the incremental drag to the total drag in the take-off and landing configuration.

TABLE XIII DATA COMPARISON			
$C_{D_{MIN}}$	at Mach 0.38	0.0214	(NOR67-30, Fig. 3.2.2)
C_{D_L}	(trimmed)	0.0167	(NOR67-30, Fig. 3.12.2)
$\Delta C_{D_{FLAPS}}$		0.0078	Table XII above
$\Delta C_{D_{GEAR}}$		0.032	Table XII above
	Total C_D	0.0779	

NOR67-30, Figure 3.15.3 presents:

C_D	Landing (free air)	0.074
C_D	Take-off (free air)	0.089

1.4 USING A THRUST MEASURING SYSTEM IN COMPUTING TOTAL DRAG

1.4.1 A thrust measuring system is useful in attaining drag data as well as the excess power characteristics of an aircraft. The flight tests did not include level flight acceleration trials for this specific purpose but a sample computation has been made in order to illustrate this procedure.

1.4.2 Level flight acceleration trials are performed by accelerating the aircraft from low airspeed to maximum airspeed at a constant altitude. If thrust is measured, then the excess power available to accelerate the aircraft may be computed from airspeed data. It is assumed that kinetic energy and potential energy are interchangeable and that the total energy is the sum of these two energies.

Total energy: $E = Wh + \frac{WV^2}{2g}$ ft. lb.

Or, specific energy: $\frac{E}{W} = E_h = h + \frac{V^2}{2g}$ $\frac{\text{ft. lb.}}{\text{lb.}}$

where:
- E = total energy, ft lb
- E_h = specific energy, ft lb/lb
- W = aircraft weight
- h = geometric altitude, ft.
- V = velocity, ft/sec
- g = gravitational constant, ft/sec^2

1.4.3 The rate of change of specific energy will be proportional to the excess power. Therefore, it is useful to compute:

$$\frac{dE_h}{dt} = \frac{dh}{dt} + \frac{V\,dV}{dt} \qquad \frac{ft.lb.}{lb.sec.}$$

1.4.4 Geometric altitude is computed from pressure altitude by correcting for non-standard densities.

$$\frac{dh}{dt} = \frac{dH_P}{dt} \cdot \frac{T_{SO}}{(T_{SO})_s} \qquad \frac{ft.}{sec.}$$

where: H_P = pressure altitude, ft.
T_{SO} = ambient static temperature, ^oR
$(T_{SO})_s$ = standard day ambient static temperature, ^oR

1.4.5 The rate of change of specific energy for standard weight, neglecting the induced drag correction due to weight, is:

$$\left(\frac{dEh}{dt}\right)_s = \frac{W}{W_s} \cdot \frac{V}{g}\,\frac{dV}{dt} + \frac{W}{W_s}\,\frac{dh}{dt}$$

where: W_S = aircraft standard weight, an arbitrary value.

1.4.6 Since only one engine was instrumented, it was necessary to assume both engines produced equal thrust. Ram drag was computed by means of the Norair gas generator method computer program. Thus, net thrust is solved as:

$$F_N = 2\,(F_G - F_{RDG}) \quad lb.$$

where: F_N = Net thrust, lb (total thrust for two engines)
F_G = Measured gross thrust, lb
F_{RDG} = Computed ram drag, lb

1.4.7 If level flight performance data are acquired, the rate of change of specific energy may be obtained from flight test results. Then, since dh/dt is zero, we may write:

$$\left(\frac{dEh}{dt}\right)_s = \frac{W}{W_S} \cdot \frac{V}{g}\,\frac{dV}{dt}$$

and $$F_N - D = \frac{W}{g}\,\frac{dV}{dt}$$

Thus: $$D = F_N - \frac{W_S}{V}\left(\frac{dEh}{dt}\right)_s = 0.7M^2\,P_{SO}\,S\,C_D$$

where: D = Drag, lb
M = Mach number
P_{SO} = ambient static pressure, lb/ft^2
S = aircraft wing area, ft^2
C_D = aircraft drag coefficient

1.4.8 As level flight acceleration performance trials were not conducted, it is only possible to illustrate the principle of computing C_D. One high altitude, short duration, full A/B, acceleration was recorded. An approximation to the level flight acceleration trial was made by using data from a portion of a takeoff flight. The latter data are inaccurate since the aircraft was intentionally climbing. Therefore, the resulting C_D data can only approximate the values reported in NOR-67-30. Data were computed by neglecting position error and assuming -35lb ejector force. The resulting data are shown in Table XIV.

1.4.9 Figure 50 illustrates the excess power characteristics as a function of Mach number for two test cases. These data were used in computing C_D values for the aircraft. NOR-67-30 data for C_D were obtained and compared with the test data in Figure 51. Minimum drag coefficient data were computed by subtracting NOR-67-30 drag due to lift data from the test result C_D data. These data are shown in Figure 52. Perfect agreement is not possible due to the many assumptions necessary in this data reduction. However, the data do agree sufficiently enough for the purposes of this demonstration.

1.5 CONCLUSIONS

1.5.1 From the above examples, it may be concluded that:

(a) Incremental drag coefficients for external stores or for various flight configurations may be determined with the aid of a thrust measuring system.

(b) A ram drag correction is relatively simple to make and will increase the precision of the resulting data.

(c) Accuracy and confidence would be greatly increased if both engines were to be instrumented with thrust measuring systems.

Table XIV: Excess Power Data

FT-02

TIME	MACH	FN	DVDT	DHDT	DEDT	DRAG	CD	CL	
0.0	.8855	3249.	4.40	-12.77	111.4	1716.	.0341	.2534	
5.7	.9137	3364.	3.89	-4.84	114.1	1843.	.0342	.2366	
11.4	.9399	3498.	3.42	1.78	108.0	2099.	.0368	.2229	
17.0	.9598	3597.	2.99	7.07	99.5	2336.	.0391	.2130	
22.7	.9797	3661.	2.61	11.05	88.3	2565.	.0413	.2049	
28.4	.9922	3696.	2.26	13.69	80.1	2715.	.0428	.2002	
34.1	.9984	3692.	1.96	15.00	75.7	2770.	.0432	.1981	
39.7	1.0026	3676.	1.70	14.98	72.5	2797.	.0434	.1969	
45.4	1.0084	3667.	1.49	13.61	68.0	2848.	.0442	.1966	
51.1	1.0138	3668.	1.32	10.90	63.6	2906.	.0448	.1955	
56.8	1.0204	3708.	1.19	6.84	57.8	3019.	.0466	.1955	
62.5	1.0441	3760.	1.10	1.45	34.1	3363.	.0502	.1887	
68.1	1.0503	3860.	1.05	-5.29	26.9	3548.	.0514	.1832	
73.8	1.0561	3915.	1.05	-13.35	19.8	3687.	.0523	.1791	
79.5	1.0632	3964.	1.09	-22.74	10.4	3845.	.0532	.1746	
85.2	1.0702	4033.	1.17	-33.44	.4	4028.	.0540	.1689	Diving
90.8	1.0731	4117.	1.30	-45.47	-4.0	4162.	.0549	.1660	
96.5	1.0760	4096.	1.47	-58.80	-8.5	4193.	.0551	.1654	
102.2	1.0798	4101.	1.68	-73.46	-14.7	4266.	.0560	.1650	

ASSUMED EJECTOR FORCE -35. LB./ENGINE

FT-03

TIME	MACH	FN	DVDT	DHDT	DEDT	DRAG	CD	CL
0.0	.3338	7481.	12.96	-8.43	148.9	2537.	.0957	.4942 *
5.7	.4060	7787.	13.80	8.25	215.1	1895.	.0484	.3335 **
11.4	.4795	8125.	14.32	22.06	274.4	1770.	.0325	.2389
17.0	.5563	8412.	14.51	33.27	329.7	1857.	.0255	.1782
22.7	.6298	8558.	14.37	41.82	375.3	1999.	.0216	.1399
28.4	.6987	8814.	13.90	47.54	408.1	2414.	.0214	.1144
34.1	.7655	9127.	13.11	50.24	425.3	3058.	.0228	.0960
39.7	.8336	9471.	11.98	49.81	420.8	3966.	.0252	.0819
45.4	.8934	9767.	10.54	46.22	391.1	4996.	.0280	.0720
51.1	.9399	10076.	8.76	39.54	346.1	6064.	.0309	.0654

ASSUMED EJECTOR FORCE -35. LB./ENGINE

* Assumed takeoff

** Assumed flaps down

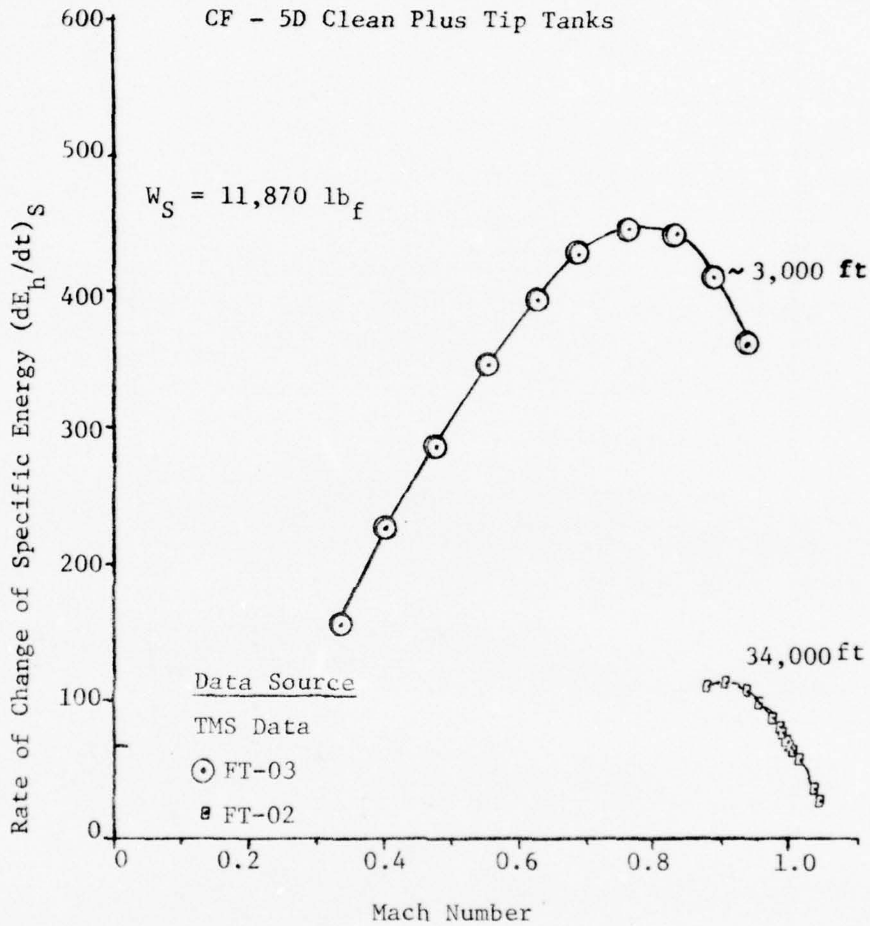

Figure 50: Excess Power Characteristics

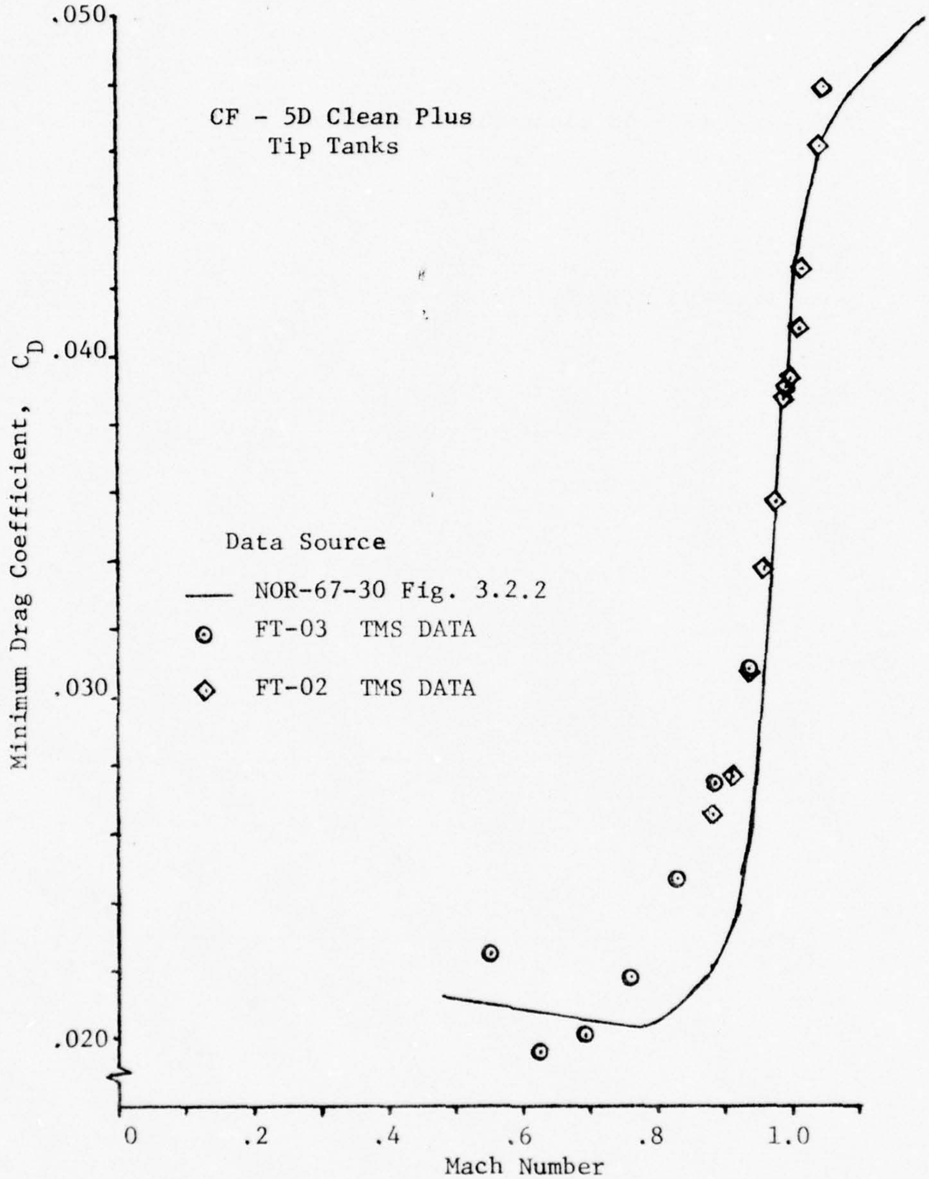

Figure 51: Minimum Drag Coefficient vs Mach Number

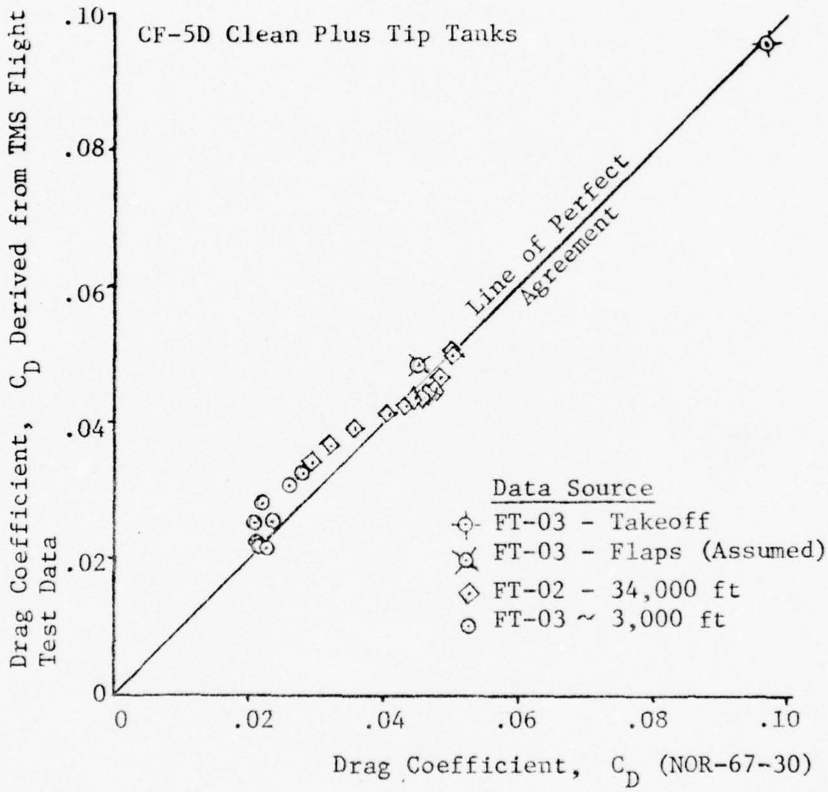

Figure 52: Drag Coefficient Derived from TMS Flight Test Data vs Basic Aerodynamic Data

APPENDIX XI

LIST OF ALL TEST EQUIPMENT

Item	Description
Transducer	NAE Static Thrust Stand for the use of Consolidated Electrodynamics Corp. Type 4-326-000, Range 0 to 250 psia, Serial No. 22329, calibrated
Transducer	NAE Static Thrust Stand for the use of Consolidated Electrodynamics Corp. Type 4-326-000, Range 0 to 250 psia, Serial No. 1260, calibrated
Amplifier	NAE Static Thrust Stand for the use of, Electro-Instruments DC Amplifier Model A-15C Serial No. A-59C
Amplifier	NAE Static Thrust Stand for the use of, Electro-Instruments DC Amplifier Model A-15C Serial No. A-3894-C
Transducer	TMS for the use of S.E. Laboratories, Feltham, U.K. Pressure transducer 0 to 10 psid Serial No. 4039 Last calibration date: April 73
Transducer	TMS for the use of S.E. Laboratories, Feltham, U.K. Pressure transducer 0 to 10 psid Serial No. 4040 Last calibration date: April 73
Transducer	TMS for the use of, S.E. Laboratories, Feltham, U.K. Pressure transducer 0 to 60 psia Serial No. 4041 Last calibration date: April 73
Transducer	TMS for the use of Conrac Corporation, Durate, Calif. Pressure transducer 0 to 60 psia Serial No. 905847, Model 4715H Last calibration date: 18 Aug 72

Item	Description
Transducer	TMS for the use of Conrac Corporation, Duarte, Calif. Pressure transducer 0 to 60 psia Serial No. 105-2, Model 4715H Last calibration date: 2 Aug 72
Transducer	TMS for the use of Conrac Corporation, Duarte, Calif. Pressure transducer 0 to 10 psid Serial No. 105-3, Model 4715H Last calibration date: 22 Mar 73
Transducer	TMS for the use of Conrac Corporation, Duarte, Calif. Pressure transducer 0 to 10 psid Serial No. 105-4, Model 4715H Last calibration date: 8 Mar 73
Transducer	TMS for the use of Conrac Corporation, Duarte, Calif. Pressure transducer -2 to +10 psid Serial No. 105-5, Model 4715H Last calibration date: 17 Aug 72
Transducer	TMS for the use of Conrac Corporation, Duarte, Calif. Pressure transducer -2 to +10 psid Serial No. 105-6 , Model 4715H Last calibration date: 1 Aug 72
Computer, thrust	TMS for the use of, ComDev type CTH1-1, Part No. 139398 Serial No. 00101 Calibrated, 20 Feb 73
Indicator, thrust (each 3)	TMS for the use of, ComDev type ITH 1-1, Part No. 139477 Serial No. 00101 and 00102 and 00103 Calibrated, 20 Feb 73

Item	Description
Pre-amplifier, temperature	Rosemount probe for the use of, ComDev, type (NIL) Serial No. (NIL) Calibrated, 19 Feb 73
Converter	RPM frequency to DC converter used in indicating aircraft engine RPM. ComDev, type (NIL) Serial No. (NIL) Calibrated: 20 Feb 73
CADC	Central Air Data Computer, CF-5D aircraft for the use of Airesearch, Part No. 949982-1-1 Serial No. 91-R1 NATO No. 6605/21-844-7991 Calibrated: 19 Feb 73 (ComDev specified modifications only calibrated)
Aircraft	Flight Test Vehicle CF-5D Tactical Fighter and Trainer Serial No. 116801
Turbojet Engine	Engine, TMS Flight Test Engine J85-CAN-15 Turbojet Engine, c/w Afterburner Serial No. 8476
Turbojet Engine	Engine, TMS Bare Engine Test Engine J85-CAN-15 Turbojet Engine, c/w Afterburner Serial No. 8611
A/B Casing Assy.	Afterburner Casing Assembly for Part No. 12A00080C01 Serial No. 5014 (Used with the flight test engine S/N 8476)
Variable Exhaust Nozzle	Part No. 605T43G02 Serial No. 5044 (Used with the flight test engine S/N 8476)

EXHIBIT A TO TEST REPORT H036/119/FR/II

The following four pages include a letter drafted by the flight testing agency; The Aerospace Engineering Test Establishment. This letter summarizes the test pilots' qualitative assessments of the ComDev thrust measuring system during flight and also proposes some future applications for the system.

DEPARTMENT OF NATIONAL DEFENCE

MINISTÈRE DE LA DÉFENSE NATIONALE

10081-71/81 (COMD)

AEROSPACE ENGINEERING TEST ESTABLISHMENT

CANADIAN FORCES BASE, COLD LAKE

MEDLEY, ALBERTA

SEP 24 1973

10081-71/81

Office of the Commander

14 September 1973

Distribution List

PILOT'S COMMENTS - COM DEV THRUSTMETER

References: A. Message PD 71/81.
B. Telecon DAES 3-3-2 and AETE FD 12 September 1973.

1. During the flight test portion of NDHQ Project Directive 71/81, 4 qualified test pilots from the Aerospace Engineering Test Establishment at CFB Cold Lake, Medley, Alberta, Canada flew a total of 18 instrumented flights in CF-5D 116801 with an operating Com Dev thrustmeter installed on the right engine. Two flights were flown in October 1972 and the remaining 16 were flown between January and March 1973. Major G. Smith (USAF) flew 14 flights, Major B. Phipps 2, and Captain J. Aitken and Major G. Henderson 1 each.

2. The following pilot comments are qualitative only and do not constitute an AETE report on the accuracy of the system. The specific accuracy of the system will be assessed on completion of a detailed data analysis.

Cockpit Thrustmeter Indicators

3. Two thrust indications are displayed on the same indicator, per cent reference thrust and gross thrust. Per cent reference thrust (reference thrust is the thrust available from a nominal engine in Military power at the existing flight conditions of airspeed/mach, pressure altitude and outside air temperature) is displayed using a rotating pointer and a scale that indicates 0% near the 11 o'clock position on the dial and increases clockwise in 2% increments to a value of 160% at the 10 o'clock position. The numbers (every 20%) on the per cent reference thrust scale of the indicator are small. This, in addition to marking every 2%, makes it possible for the pilot to misread the value on the indicator.

4. Gross thrust, measured in pounds, is displayed digitally in the centre of the dial using four digits with the units' digit frozen at zero. The numbers are large enough to be easily readable. During rapid throttle movements the tens' digit is unreadable and even the hundreds' digit is hard to read due to its rapid change. At stabilized power settings (even with aircraft acceleration or deceleration) the tens' digit varies slowly enough to be readable. Three pilots consider the use of a frozen units' digit to be a good choice while one also desires the tens' digit to be frozen at zero.

Comments On Operation of the Thrustmeter

5. After engine start, with the aircraft at rest and the right engine in idle, the thrustmeter registered small values of both gross thrust and per cent reference thrust. Since the pilots had little or no previous experience with jet engine idle thrust figures, they noted the values but did not compare them to any expected value. Advancing and retarding the throttle while taxiing caused the indicated thrust value to increase and decrease with little apparent lag. At a fixed throttle setting, the per cent reference thrust pointer remained steady and only the tens' digit of the gross thrust indicator fluctuated (approx ± 10 lb).

6. While advancing the throttle to military power prior to take-off, thrust readings increased consistently with the apparent thrust increase. At stabilized military power, with the engine auxiliary air doors closed, the gross thrust and the per cent reference thrust stabilized. When the engine auxiliary air doors were opened, the gross thrust indication increased close to the amount expected (approx 300 lb thrust per engine). The gross thrust indicated slightly below the listed military thrust in the aircraft technical order (2952 lb for a sea level static, uninstalled bare engine). With the opening of the auxiliary air doors the per cent reference thrust increased approximately 14% and indicated approximately 100% in October 1972 and approximately 94% in January 1973. Turning the engine anti-ice system on and off caused a reduction and increase in thrust close to that listed in the aircraft technical order (9% of military thrust).

7. For maximum afterburner take offs, maximum afterburner was selected shortly after brake release. Both thrust indications increased and seemed to follow the increase in thrust. The gross thrust in maximum afterburner initially showed a value close to that listed in the aircraft technical order (4,300 lb for a sea level static, uninstalled bare engine). The per cent reference thrust increased initially to the 140% range.

8. During takeoff and level acceleration at low altitude, the gross thrust indication increased continuously with increasing indicated airspeed, whether in Military or Maximum afterburner. During acceleration in Maximum afterburner from minimum to maximum airspeed, the per cent reference thrust increased as much as 15%. During acceleration in Military power, the per cent reference thrust remained near 100% but did increase slightly (a maximum increase of approximately 4 per cent).

9. During flight the gross thrust indication increased with increasing airspeed at constant altitude and decreased with constant mach and increasing altitude. During inflight engine shutdown both thrust indications remained near their low idle settings. The thrustmeter did not positively indicate engine shutdown or relight.

10. During the flight test program, a very brief simulation of the drag of different external store configurations was carried out using landing gear, flaps, and speed brakes. Changing thrust on only the instrumented (right) engine, the pilot recorded the thrust required for a number of different configurations at the same stabilized altitude and airspeed. The increases in thrust required were consistent for the different configurations. Example: The thrust required was noted for a clean aircraft, for an aircraft with full flaps extended, for an aircraft with speed brakes extended, and for an aircraft with both full flaps and speed brakes extended. The increase in thrust required for extending the full flaps PLUS the increase for extending the speed brakes APPROXIMATELY EQUALLED the increase for extending both at the same time.

Potential Uses of a Thrustmeter

11. After a high degree of accuracy has been proven, the following would apply:

a. All four test pilots agreed that a thrustmeter would be valuable during an engine check just prior to take off. They definitely desired some indication of engine thrust for this check. One pilot with considerable EPR (Exhaust Pressure Ratio) gauge experience believed an EPR gauge gave equally valuable information. The other three pilots, two with considerable EPR gauge experience, strongly preferred a thrustmeter.

b. During test or functional check flights the engine operation can be rapidly checked throughout the flight envelope, dealing with unsubstantiated pilot comments such as a particular aircraft 'felt underpowered'. Modifications to aircraft engines can be easily assessed inflight with regard to their affect on thrust output. Drag measurements on new aircraft or new aircraft configurations, including external stores, can be rapidly carried out throughout the flight envelope.

c. For an operational pilot during flight, the raw data displayed on the thrustmeter would appear to be of use only as an aid in selecting cruise, climb, or descent power settings.

However, if the thrustmeter data are converted to such information as thrust-to-weight ratio, excess thrust (thrust minus drag), or specific energy, it may be possible to devise some kind of display which would show a pilot an optimum manner in which to manoeuvre his aircraft.

L.H. Keelan
Colonel
Commander, AETE

DISTRIBUTION LIST

Action	Information
NDHQ/DAES 3-3-2	NDHQ/DAMR-6

EXHIBIT B TO TEST REPORT H036/119/FR/II

SPECIAL REPORT

PROBABLE ERROR STUDY OF THRUSTMETER MONITORING SYSTEM

H036/120G

SUBMITTED TO

MR. J.R.B. MURPHY
MANAGER, RESEARCH DEPARTMENT
COMPUTING DEVICES OF CANADA LIMITED

5 JULY 1973

SUBMITTED BY

MR. A. GRAVELLE
ENGINEER, RESEARCH DEPARTMENT

AETE CONSULTANT:

CAPTAIN B. TURNER
ENGINEER,
AVIONICS SECTION

TABLE OF CONTENTS

1. SUMMARY

1.1 A study has been made of the system probable errors in acquiring, recording and computing data for the thrust measuring project flight trials. Errors arising from manufacturer's tolerances, calibration and least readings have been considered. The method of probable errors has been used to produce the probable errors in each recorded parameter.

1.2 It was concluded that:

(1) With the exception of Mach No., exhaust gas temperature, maximum fuel flow, compressor static pressure and power lever angle, all recorded data are within desired tolerances. Mach No. only marginally exceeds desired limits.

(2) Thrusts computed by the thrust measuring system (TMS) and by the data processing computer using TMS input data may have probable errors (PEs) of ±0.11% of computed gross thrust or ±0.09% of percent reference thrust. Maximum differences may be double the PE's.

1.3 It was not practical to conduct extensive testing in order to determine precise probable error data for this report. Many of the errors used are based upon opinion and past experience. They are included as a record of the possibility of error due to the circumstances for which the estimates were made.

2. BACKGROUND

2.1 A ComDev designed and developed thrust measuring system (TMS) for jet engines has been flight tested in a Canadian Defence Forces CF-5D aircraft. The aircraft was supplied and rigged for testing by the Aerospace Engineering Test Establishment (AETE) at CFB Cold Lake, Alberta.

2.2 AETE installed ComDev designed probes in the starboard engine of aircraft SN801. A ComDev TMS computer was installed in the rear seat well. TMS indicators were installed in the forward cockpit and in a special photo-panel in the rear cockpit. The photopanel also contained a set of flight instruments.

2.3 The TMS and 14 aircraft parameters were monitored continuously by a data acquisition system. Data were recorded by a digital tape recorder located in the rear cockpit. Two data acquisition rates were used:

(1) TMS input and output data - 2 readings/sec.

(2) Aircraft data - 1 reading/3 sec.

2.4 Data processing was accomplished by means of an IBM System /360/30 computer in Edmonton and a CDC 6400 computer at ComDev.

3. METHOD OF ANALYSIS

3.1 It is a common engineering practice to state the degree of uncertainty associated with any absolute measurement. Uncertainties may arise with the method of measurement, precision of recording, etc. Every measurement could be stated along with some estimate of the uncertainty of its accuracy. This is accomplished by estimating the uncertainty such that 50% of the time the absolute value will be equal to the measured value plus or minus the uncertainty. A 50% level of uncertainty is called the probable error (PE) of a measurement.

3.2 It is possible that a number of PEs could accumulate and result in a large total uncertainty. The likelihood of this happening is probably small. Therefore, it is usual to solve the root of the sum of the squares of PEs. The resulting total PE will exceed the largest individual PE but will be smaller than the absolute sum of all the PEs. The PE method has been used in this report.

3.3 Data were sensed by means of pressure transducers, temperature probes and variable resistors. Outputs from these sources are voltages which are converted by the recorder or the TMS to binary code. Binary coded data are stored on magnetic tape. A data processing computer is used to read the data tapes and compute engineering data. Binary code is converted to engineering data by means of calibration data. Probable errors may be introduced in acquiring, converting and storing data.

3.4 Error sources have been examined and probable errors have been estimated wherever possible. The error sources may be defined as follows:

1. Analog to digital (A/D) conversion tolerance.
2. Truncation (precision) in recording data.
3. Position error in the Central Air Data Computer.
4. Manufacturing tolerance in temperature probes and pressure transducers.
5. Hysteresis observed in calibrating transducers.
6. Faulty signal conditioning in the exhaust gas temperature circuit.
7. Calibration method.
8. Data reduction method.

(1) Item 6 does not impact the thrust measuring system.

3.5 A few measurements were made in order to provide information concerning flight operating conditions. These data do not influence the TMS. Probable errors should be expected due to the following reasons but these PEs have not been estimated.

1. Electrical systems used to measure aircraft parameters have not been rigorously tested to determine their stability under a wide range of operating conditions. It is possible that ambient conditions could influence the precision of these data.

2. Indicated pressure altitude, Hp, and airspeed, Vi, are subject to position error. The extent of this error has not been estimated.

3. Fuel flow meters may be in error if the fuel specific gravity varies from that of the calibration fluid. This error will depend upon fuel temperature.

4. Fuel remaining data may be in error if the fuel quantity indicating system is in error.

4. RESULTS - PROBABLE ERROR ANALYSIS

4.1 A/D Conversion

4.1.1 Two A/D converters are used; one in the TMS and one in the recorder. The TMS converter is considered to be capable of resolving data to a precision of ±1 count in 4095. This converter is used in recording T_{T_i}, Mach, P_{so}, ΔP, ΔP_s and P_{s7}. The TMS A/D converter probable error is:

$\pm(1/4095) \times$ Full Scale Reading

4.1.2 The recorder A/D converter is not the same type as the TMS uses. Recorder A/D conversions are considered to have a PE of ±3.5 counts in 4095. Thus, the recorder A/D converter PE is:

±(3.5/4095) x Full Scale Reading

4.2 Recording Precision

4.2.1 Measured data are truncated by the recorder and written on tape. Truncation is not a factor of A/D performance. Truncation is a necessity resulting from the fact that recording must be done in a fixed format. TMS data are recorded as 12 bit integers and aircraft data are recorded as 11 bit integers. It is assumed that the uncertainty of recording any value will be one half the recording least reading. Therefore, the PE of recording will be as follows:

12 bits = integer 4095

11 bits = integer 2047

FSR = full scale reading

TMS data PE = ±0.5 x 1/4095 x FSR = ±0.000122 FSR

Aircraft data PE = ±0.5 x 1/2047 x FSR = ±0.000244 FSR

4.3 Central Air Data Computer (CADC)

4.3.1 The aircraft CADC is used to provide ambient total temperature ($\sqrt{T_{Ti}}$), Mach No. and ambient static pressure data. The CADC corrects measured data for pitot static errors which are due to aircraft position. AETE trials have indicated the presence of a small residual position error as indicated in Figure 53. This graph indicates an altimeter position error correction of 75 ft. at sea level and Mic[1] of 0.8. A 75 ft. correction may be extrapolated to altitude as shown in Table XV by means of flight test position error tables.

---

(1) Mic is the instrument corrected Mach number.

ALTITUDE	SL	20000	36000 FT
Hpc(ft)	75	65	56
Mpc	0.003	0.003	0.003
Pspc(psi)	-0.04	-0.02	-0.01

Table XV - Position Error Corrections at Mic 0.8

4.3.2 Since position error corrections are not always needed, a PE of half the position error correction will be assumed reasonable. Therefore, Mach will have an assumed PE of ±0.0015 and Pso will have an assumed PE of ±0.01 psi.

4.3.3 The total temperature will be in error if air is not flowing over the temperature probe. Therefore, data acquired while the aircraft is on the ground may be error. This error is not included as a PE. Manufacturer's data have been examined and it was concluded that a PE of ±1.5°R is reasonable for the ambient temperature data.

4.3.4 Mach data from the CADC are only valid within the range 0.17 to 1.6. The manufacturer indicated that data within this range will have a PE of ±0.012.

4.4 SE Transducers

4.4.1 SE transducers are used in measuring engine pressures. The transducers were calibrated over their range of 0 to 1.0 volts. They were found to have small deviations in linearity as shown in Figure 54. Calibration corrections are not made by the TMS. PEs of half the deviation range and a hysteresis of about 0.5 millivolts will be assumed in this report. The curves indicate millivolt PEs as shown in Table XVI.

4.4.2 The manufacturer has stated that errors will arise due to g-loading, temperature and mechanical hysteresis. These errors have been examined and it was concluded that a PE of ±0.5% full scale should be accepted. These PEs are shown in Table XVI.

POSITION ERROR CALIBRATION

ALTITUDE POSITION ERROR CORRECTION vs INSTRUMENT CORRECTED MACH NUMBER

BASELINE LOADING

(ANNEX F, PROJECT 71/66, FIGURE F-1)

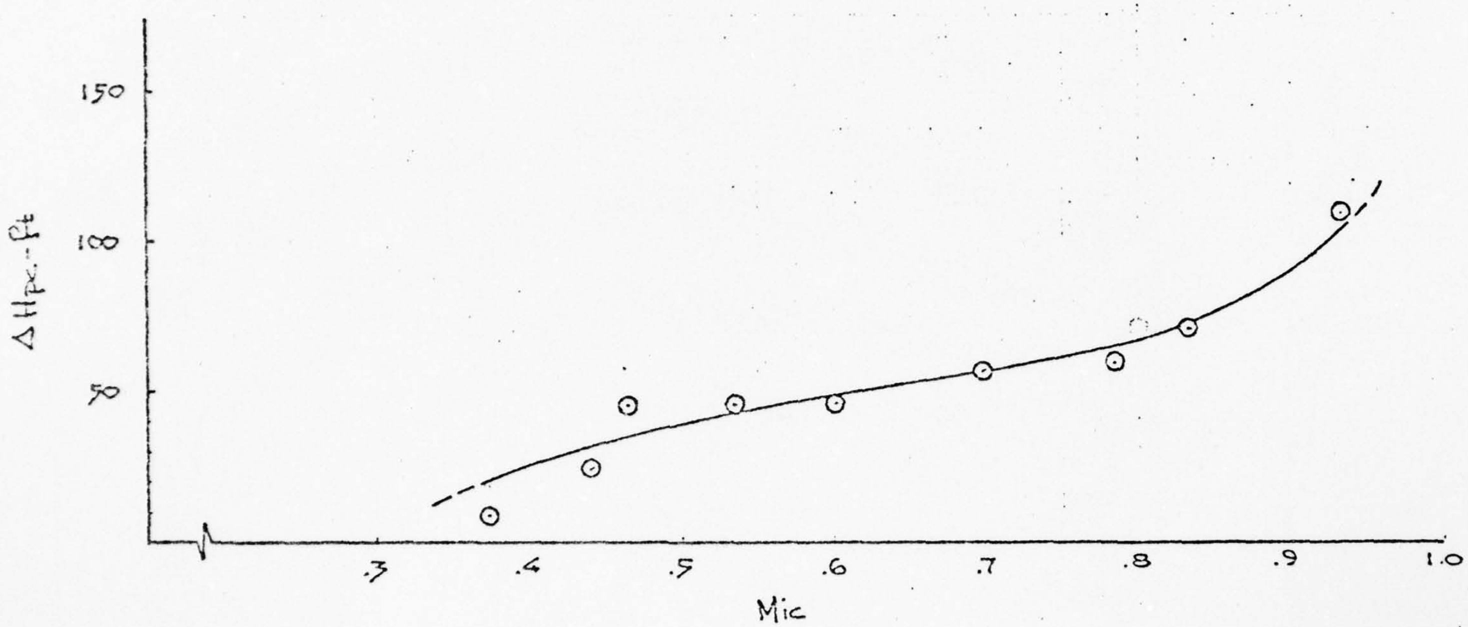

Figure 53: Position Error Calibration

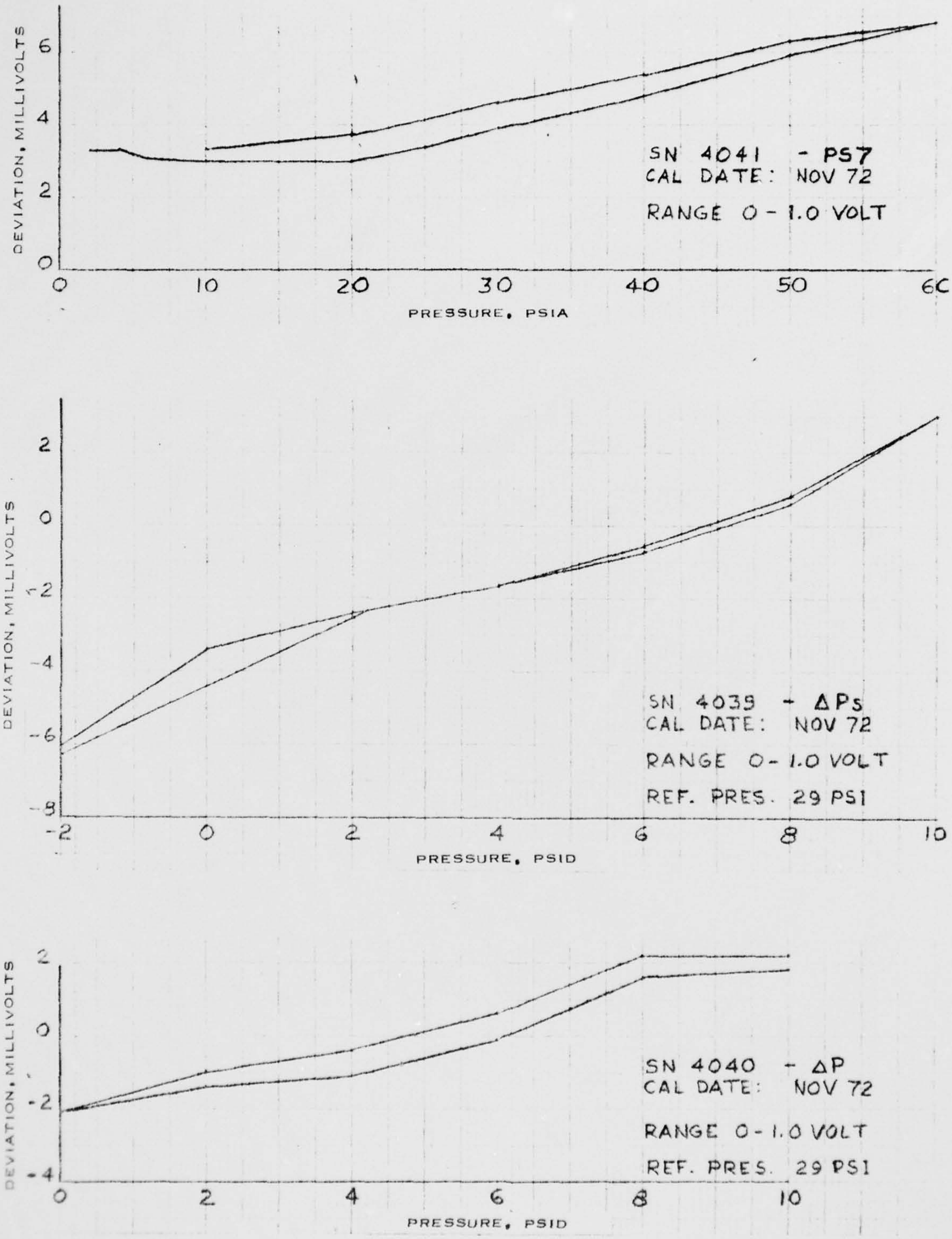

Figure 54: SE Transducer Calibration

PARAMETER	FULL SCALE	PE CALIBRATION	PE HYST.	PE MANUF.
Ps7	0-60 psia	± 2 mv	± 0.5 mv	± 0.3 psi
ΔPs	-2 to 10 psid	4.5 mv	0.5 mv	0.06 psi
ΔP	0-10 psid	2.0 mv	0.5 mv	0.05 psi

Table XVI: PEs in SE TRANSDUCERS

4.5 Aircraft Parameters

4.5.1 General Comments

4.5.1.1 Parameters indicative of engine control settings and aircraft operating conditions are monitored and recorded on tape. Aircraft parameters were calibrated by AETE. Calibration curves are attached as Appendix A.

4.5.1.2 PE data were estimated by considering information provided by manufacturers, AETE engineer's assessment and by examining calibration curves. PEs due to circuit stability and ambient conditions have not been experimentally determined. PE estimates include conservative predictions of the effects on gain and drift of signal conditioners for given ambient conditions.

4.5.2 Exhaust Gas Temperature T_{T5H}

4.5.2.1 Exhaust gas temperatures recorded during flight trials were observed to be scattered much more than could be due to normal PE. Figure 55 a plot of typical steady state data acquired during flight testing. The PE for this data set is ±17.8°C. It is believed that the signal condition circuit was at fault since both the reference junction and the amplifier were changed more than once. The system calibrated well on the ground.

4.5.3 Position Error in Altitude and Airspeed

4.5.3.1 Altitude and airspeed are sensed by transducers in the aircraft pitot system. These parameters are subject to position error which will be a function of airspeed and altitude. Since these parameters are used only as an indication of flight conditions, position errors were not included in this study for these two parameters.

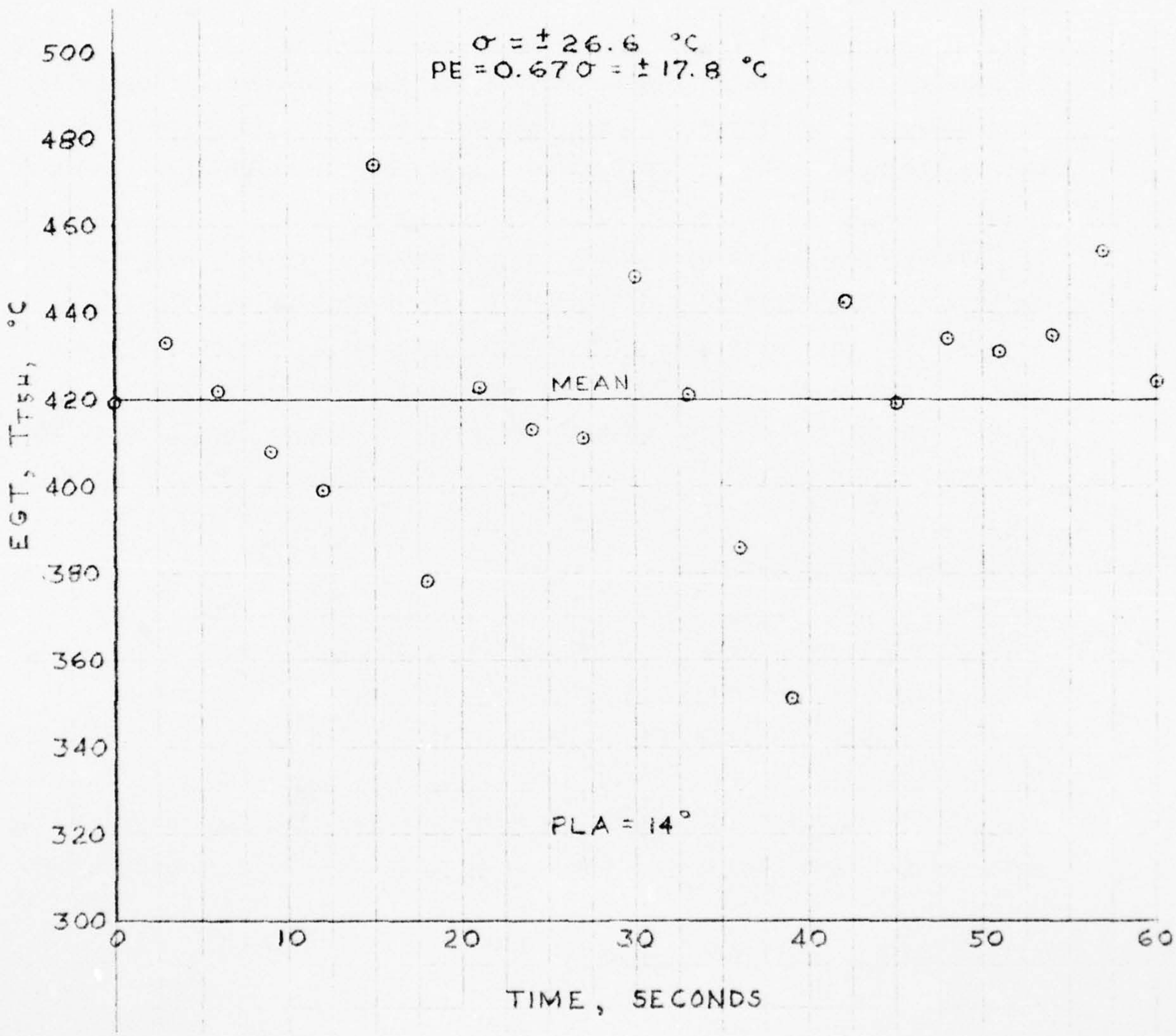

Figure 55: Data Scatter Recorded in T_{T6H}

4.5.4 Fuel Flow and Fuel Remaining

4.5.4.1 Fuel flow turbines are calibrated at 80°F with MIL Spec Type II fluid having the specific gravity of 0.770 at 80°F. A correction should be made for the actual specific gravity of the fuel. This error source is not included as a PE.

4.5.4.2 The fuel flow turbines were calibrated and rated at ±2% of indicated flow. Fuel flow may reach 3,000 lb/hr for the main and 5,000 lb/hr for the afterburner at maximum engine thrust. This calibration error will exceed the desired ±50 lb/hr at maximum flows.

4.5.4.3 The fuel remaining system is temperature compensated and calibrated by the manufacturer such that the shape of the fuel tank is accounted for. No estimate of the residual PE due to tank shape or fuel temperature will be included here.

4.5.5 Ejector Static Pressure Psej

4.5.5.1 Psej is measured by means of four static ports located on the ejector wall and connected by a common pipe to a transducer. Ejector static pressure is by nature a function of the measurement location. It was not possible to experiment with the location of the ports. AETE attempted to locate the ports in the same position as NORAIR did during previous in-flight thrust measurement trials. A PE due to the location of the static ports will not be included in this report.

4.5.5.2 The Psej transducer is subject to error due to temperature and g-loading. The following PE estimates are based upon manufacturer's specifications.

(1) Temperature. The transducer was calibrated at 72°F and rated for operation between 0°F and 450°F. As actual temperature data were not available, an arbitrary maximum temperature of 150°F was assumed. The transducer is rated at ±0.005% of full scale per °F and full scale is 20 psia. Therefore, the resulting PE is ±0.088 psi.

(2) g-loading. The transducer is rated at ±0.08% full scale per g. The tail section is estimated to be subject to a ±5g vibration loading at the transducer location. Therefore, the resulting PE is ±0.08 psi.

(3) AETE engineers have estimated that a calibration error of ±0.05 psi and an amplifier error of ±0.05 psi are also present.

4.5.6 Compressor Discharge Pressure, Ps3

4.5.6.1 A static port is located in the bleed air line from the engine compressor. The transducer used in measuring Ps3 is rated at ±1.5% of indicated pressure. In-flight pressures range from 50 to 100 psia. A mean value of 75 psia was used in estimating the PE due to manufacturer's tolerance.

4.5.7 Power Lever Angle, PLA

4.5.7.1 Power lever angle settings have to be within tolerances specified in EOs. An engine technician sets the PLA at idle (12 to 15 degrees) and at MIL (90 to 93 degrees). Intermediate points are determined by linear interpolation. An A/B point was obtained from test cell data during functional tests. The power lever position is sensed by a variable resistor. AETE engineers estimate that their calibration technique includes a PE of ±1.5 degrees.

4.5.7.2 It is estimated that the mechanical linkage system will introduce a mechanical hysteresis type of PE in the order of $\pm 2^{\circ}$.

4.5.8 Nozzle Position Indicator, NPI, and Area A8

4.5.8.1 The nozzle position indicator is a three phase synchro system used to indicate the variable exhaust nozzle (VEN) position. Voltage levels for the three phases were monitored and recorded on tape. An algorithm was devised to determine a unique nozzle position corresponding to the three measured voltages.

4.5.8.2 Nozzle areas, A_8, were measured by setting the NPI at desired points, measuring phase voltages and pressing a foam sheet against the VEN area. The foam sheet was used to trace an outline on paper. A planimeter was used to measure the traced area in square inches.

4.5.8.3 Nozzle position indicator data are estimated to have a PE of $\pm 0.52^\circ$. This induces a PE in A_8 of ± 0.5 sq. in. A hysteresis type of uncertainty of ± 2 sq. in. was observed in the A_8 calibration curve. Therefore, a PE of ± 2 sq. in. was assumed to be sufficient to include hysteresis and errors due to positioning the nozzle and measuring areas.

4.5.8.4 Corrections were not made for pressure loading or temperature affects on A_8.

4.6 Computed Data

4.6.1 The TMS computer was monitored during flight trials by recording the following data.

(1) Input data to the TMS computer

(2) TMS computed gross thrust

(3) TMS computed reference thrust

4.6.2 TMS performance is judged by using the TMS input data in a data processing computer and solving thrust by the TMS equations. TMS computed thrust should be equal to the data processing computer results except for PEs due to recording the input and computed data.

4.6.3 Since the TMS computer and the data processing computer are not using identical input data, their computed results will not be equal. The PEs due to least readings will be used as a means of estimating the difference which should be expected. Least readings and PEs for the input data are shown in Table XVII.

DATA	RANGE	LEAST READING	PE
T_{Ti}	392 to 780^oR	±0.1 ^oR	±0.05 ^oR
Mach	0.1 to 1.7	0.0004	0.0002
Pso	1.687 to 15.236 psia	0.003 psi	0.0015 psi
ΔPs	-2 to 10 psia	0.003 psi	0.0015 psi
ΔP	0 to 10 psia	0.002 psi	0.001 psi
Ps7	0 to 60 psia	0.02 psi	0.01 psi

Table XVII- PE in TMS Input Data Recording

4.6.4 Three sample computations were made in order to estimate the PE in using recorded data to solve thrust. Data were selected from test results of Flight Test FT-02, 30 Jan. 73. These data are shown in Table XVIII.

4.6.5 The TMS input data of Table XVIII were used to solve gross and reference thrusts. Input data were then perturbed by the PEs listed in Table XVII and thrusts were computed again. The resulting errors in pounds and % of computed thrust were solved and listed in Table XIX.

4.6.6 Data shown in Table XIX indicate the magnitude of error to be expected between the TMS computer and the data processing computer due to PEs in recorded input data. Assuming that PEs are no worse than the largest errors in Table XIX, PEs of the following order should be expected.

(1) ±0.074% gross thrust

(2) ±0.051% reference thrust pounds

(3) ±0.090% % reference thrust

4.6.7 TMS computed gross and reference thrusts are truncated prior to recording. The least readings and PEs for these data are shown in Table XX.

TIME	ALTITUDE	REF.T.	MACH	T_{Ti}	Pso	Δ P	Δ Ps	Ps7
13:21:19.7	2,200 ft	100%	0.4720	496.03 ^{o}R	13.33 psia	4.364 psid	-0.042 psid	33.22 psia
13:20:42.3	2,200	100	0.2296	481.27	13.52	3.995	-0.028	30.43
13:39:00.9	33,500	135	0.9922	474.83	3.68	2.061	2.636	11.47

Table XVIII-Sample TMS Input Data

GIVEN DATA / COMPUTED DATA

MACH	TT	PSO	PT5PS6	PS6PS7	PS7	FG	RT
.4720	496.03	13.3300	4.3640	-.0420	33.22	3375.0	3410.7

DATA ERROR

MACH	TT	PSO	PT5PS6	PS6PS7	PS7	FG	ERROR	ER PC	RT	ERROR	ER PC
.0002	0.00	0.0000	0.0000	0.0000	0.00	3375.0	0.0	0.00	3411.4	.7	.02
0.0000	.05	0.0000	0.0000	0.0000	0.00	3375.0	0.0	0.00	3410.4	-.3	-.01
0.0000	0.00	.0015	0.0000	0.0000	0.00	3374.8	-.2	-.01	3411.1	.4	.01
0.0000	0.00	0.0000	.0010	0.0000	0.00	3375.2	.3	.01	3410.7	0.0	0.00
0.0000	0.00	0.0000	0.0000	.0015	0.00	3375.7	.8	.02	3410.7	0.0	0.00
0.0000	0.00	0.0000	0.0000	0.0000	.01	3376.1	1.1	.03	3410.7	0.0	0.00

GIVEN DATA / COMPUTED DATA

MACH	TT	PSO	PT5PS6	PS6PS7	PS7	FG	RT
.2296	481.27	13.5200	3.9950	-.0280	30.43	2943.7	2934.6

DATA ERROR

MACH	TT	PSO	PT5PS6	PS6PS7	PS7	FG	ERROR	ER PC	RT	ERROR	ER PC
.0002	0.00	0.0000	0.0000	0.0000	0.00	2943.7	0.0	0.00	2934.9	.3	.01
0.0000	.05	0.0000	0.0000	0.0000	0.00	2943.7	0.0	0.00	2934.3	-.3	-.01
0.0000	0.00	.0015	0.0000	0.0000	0.00	2943.5	-.2	-.01	2934.9	.3	.01
0.0000	0.00	0.0000	.0010	0.0000	0.00	2944.0	.3	.01	2934.6	0.0	0.00
0.0000	0.00	0.0000	0.0000	.0015	0.00	2944.4	.7	.02	2934.6	0.0	0.00
0.0000	0.00	0.0000	0.0000	0.0000	.01	2944.9	1.1	.04	2934.6	0.0	0.00

GIVEN DATA / COMPUTED DATA

MACH	TT	PSO	PT5PS6	PS6PS7	PS7	FG	RT
.9922	474.83	3.6800	2.0610	2.6360	11.47	2516.6	1868.4

DATA ERROR

MACH	TT	PSO	PT5PS6	PS6PS7	PS7	FG	ERROR	ER PC	RT	ERROR	ER PC
.0002	0.00	0.0000	0.0000	0.0000	0.00	2516.6	0.0	0.00	1869.0	.6	.03
0.0000	.05	0.0000	0.0000	0.0000	0.00	2516.6	0.0	0.00	1868.3	-.1	-.01
0.0000	0.00	.0015	0.0000	0.0000	0.00	2516.3	-.3	-.01	1869.2	.8	.04
0.0000	0.00	0.0000	.0010	0.0000	0.00	2516.8	.2	.01	1868.4	0.0	0.00
0.0000	0.00	0.0000	0.0000	.0015	0.00	2517.2	.5	.02	1868.4	0.0	0.00
0.0000	0.00	0.0000	0.0000	0.0000	.01	2518.4	1.8	.07	1868.4	0.0	0.00

Table XIX: Typical Thrust PEs

4.6.8 Percentage errors are solved for the gross thrust and % reference thrust computations. The combined PEs due to thrust least reading and TMS input data recording will cause errors in this computation as follows at the 2,518 lb gross thrust level and 1868 lb reference thrust.

(1) ±0.11% gross thrust error computation

(2) ± 0.055% reference thrust pounds

(3) ± 0.092% percent reference thrust error computation.

4.7 A table of parameters and their PEs has been included as Table XXI The total PE shown in the right hand column is the square root of the sum of the squares of the other errors.

THRUST	LEAST READING	PE
Gross Thrust LB.	± 3.9 lb	± 2 lb
% Reference Thrust	± 0.04%	±0.02 %

Table XX - PE in Recording Thrust

Table XXI

PARAMETER	HYSTERESIS	MANUFACTURE	CALIBRATION	AD CONV.	RECORDING	P.E.
Compressor discharge static pressure Ps3 (psia)		1.1	0.20	0.13 (1)	0.037 (2)	1.13
Power lever angle (degrees)	2.1		1.5	0.10	0.029	2.58
Exhaust gas temp. T_{T5H} (^{o}C)	0.2	9.	15.3 (3)	0.86	0.24	17.8
Main fuel flow (lb/hr)	1.1		±2%	2.9	0.83	3.2 ±2% (7)
A/B fuel flow (lb/hr)	2.2		±2%	6.4	1.8	7.0 ±2% (7)
RPM (%)			0.5	0.25	0.07	0.56
Ejector static pressure Psej (psia)		0.088 0.08	0.05 0.05	0.017	0.005	0.14
Nozzle position NPI (degrees)			0.5	0.16	0.02	0.52
Nozzle area A8 (in^2)	2		0.5			2.1 (4)
Altitude Hp (ft)	4			34	9.8	36. (5)
Airspeed Vi (kt)				0.73	0.21	0.76 (5)
Fuel remaining (lb)				0.73	0.21	0.76 (6)
Amb Air temp T_{T_i} (^{o}R)		1.5		0.05	0.05	1.5
Mach No. $0.17 \leq M \leq 1.6$		0.012	0.0015	0.0002	0.0002	0.012
Amb static pressure **Pso (psia)**		0.076	0.01	0.0015	0.0015	0.077

[illegible] (psia)		0.08	0.05			
Nozzle position NPI (degrees)			0.5	0.16	0.02	0.52
Nozzle area A8 (in^2)	2		0.5			2.1 (4)
Altitude Hp (ft)	4			34	9.8	36. (5)
Airspeed Vi (kt)				0.73	0.21	0.76 (5)
Fuel remaining (lb)				0.73	0.21	0.76 (6)
Amb Air temp T_{T_i} (^oR)		1.5		0.05	0.05	1.5
Mach No. $0.17 \leq M \leq 1.6$		0.012	0.0015	0.0002	0.0002	0.012
Amb static pressure Pso (psia)		0.076	0.01	0.0015	0.0015	0.077
Ps6-Ps7 (ΔPs) (psid)	0.006	0.06	0.054	0.0015	0.0015	0.081
Pt5-Ps6 (ΔP) (psid)	0.005	0.05	0.020	0.001	0.001	0.054
Ps7 (psia)	0.030	0.30	0.12	0.007	0.007	0.32
Recording gross thrust (lb)					2.	2.
Recording % ref. thrust (%)					0.02	0.02
Computing gross thrust error (%) at 2518 lb gross thrust						0.11
Computing % ref. thrust error (%) at 135% ref thrust						0.092

NOTES

(1) RECORDER A/D PE ±3.5/4095 COUNTS
TMS A/D PE ±1.0/4095 COUNTS

(2) AIRCRAFT DATA RECORDING PRECISION 1/2047
TMS DATA RECORDING PRECISION 1/4095
PES ARE 1/2 RECORDING PRECISION

(3) PE DUE TO NOISE IN ELECTRICAL CIRCUIT

(4) AREA IS NOT CORRECTED FOR TEMPERATURE RISE OR PRESSURE LOADING DURING ENGINE OPERATION

(5) POSITION ERROR NOT INCLUDED

(6) FUEL SYSTEM ERROR NOT INCLUDED

(7) FUEL FLOW TURBINE ACCURACY IS ±2 PER CENT OF INDICATED FLOW

Table XXI: Probable Error Summary

5. CONCLUSIONS

5.1 A desired level of accuracy for the data acquisition system was reported in the ComDev Monthly Report H031/MR37, dated 11 Sept. 72. An extract of this report and the PEs resolved herein are shown in Table XXII. Table XXII indicates that, with the exception of the following, all PEs are within desired limits.

(1) Exhaust gas temperature
(2) Power lever angle
(3) Compressor discharge pressure
(4) Fuel flow at maximum flow rates
(5) Mach No.

5.2 The exhaust gas temperature error is believed to be caused mainly by a faulty signal conditioner.

5.3 Mechanical hysteresis and the calibration method cause the PE in the power lever angle to exceed limits.

5.4 The compressor static pressure error is large due to the location of the static port and the precision of the transducer used.

5.5 The Mach No. PE is marginally greater than the limit due to the manufacturer's tolerance.

5.6 Differences in computed thrusts obtained from the TMS and from the data processing computer are solved as system errors. This computation will be in error due to recording PEs. Maximum errors will probably not exceed double the following PEs.

(1) ±0.11% in gross thrust error computation
(2) ±0.092 percent reference thrust error computation.

5.7 Special care should be taken when using altitude, fuel flow and nozzle area data as certain corrections have not been made for known error sources.

5.8 It was not practical to conduct extensive testing in order to determine precise PE data for all the factors considered in this report. Many of the PEs used are based upon opinion and past experience. They are included as a record of the possibility of error due to the circumstances for which the estimates were made.

AD-A040 092 COMPUTING DEVICES CO OTTAWA (ONTARIO) F/G 21/5
EVALUATION OF AN AIRBORNE THRUST COMPUTING SYSTEM. VOLUME I. SY--ETC(U)
MAY 75 J A GRAVELLE F33657-69-C-0733
UNCLASSIFIED H036/119/FR/I ASD-TR-75-2-VOL-1 NL

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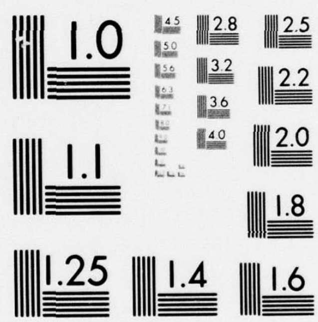


MICROCOPY RESOLUTION TEST CHART
NATIONAL BUREAU OF STANDARDS-1963-A

5.9 The nozzle position indicator was monitored by measuring the voltages on the three phase synchro. VEN areas, A_8, were inferred from the NPI data. Corrections for pressure loading and temperature effects were not included in computing A_8 data.

VARIABLE	DESIRED ACCURACY (1)	ESTIMATED P.E.
	±	±
Compressor discharge static pressure	0.5 psi	1.1 psi
Power lever angle	0.5 degree	2.6 degree
Exhaust gas temperature	$0.5^{o}C$	$17.8^{o}C$
Main fuel flow	50 pph	3.2pph±2% pph
A/B fuel flow	50 pph	7.0 pph±2%pph
RPM	0.5%	0.56%
Ejector static pressure	-	0.14 psi
Nozzle position	1 degree	0.5 degree
Pressure altitude	50 ft	36 ft
Indicated airspeed	2 kt	0.76 kt
Fuel remaining	50 lb	1.9 lb
Outside Air Temperature	$2^{o}R$	$1.5^{o}R$
Mach No. ($0.17 \leq M \leq 1.6$)	0.01	0.012
Outside air pressure	-	0.077 psi
ΔPs	-	0.081 psi
ΔP	-	0.054 psi
Ps7	-	0.32 psi

(1) Reference: ComDev Monthly Report h031/MR37, 11 Sept. 72

TABLE XXII: DESIRED ACCURACY AND PE IN RECORDED DATA

Appendix A
to Special Report
dated 15 March 1973

Calibration Data

Compressor Static Pressure

Power Lever Angle

Exhaust Gas Temperature

Main Fuel Flow

A/B Fuel Flow

%RPM

Ejector Static Pressure

Nozzle Position

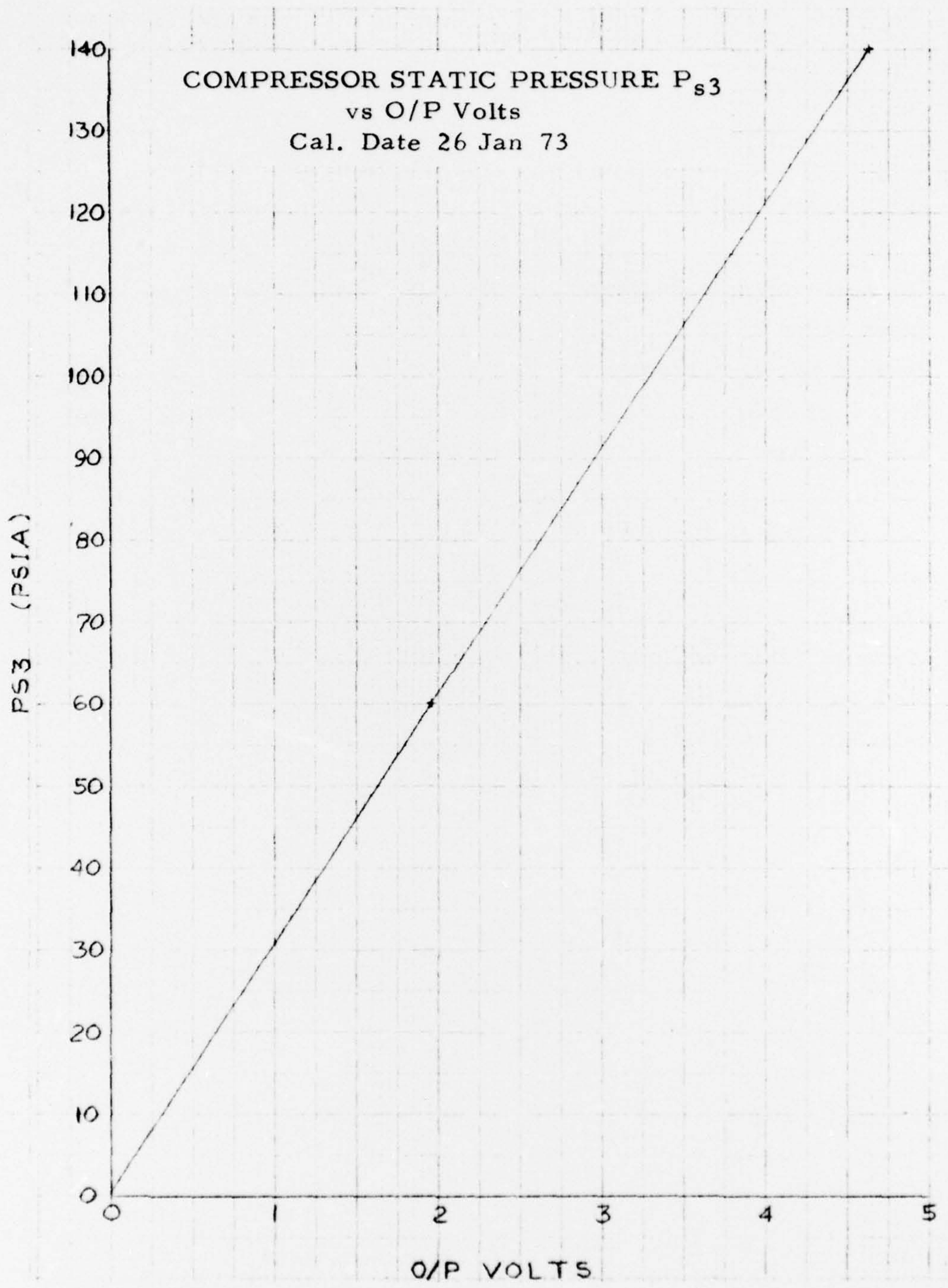
Area A_8

Pressure Altitude

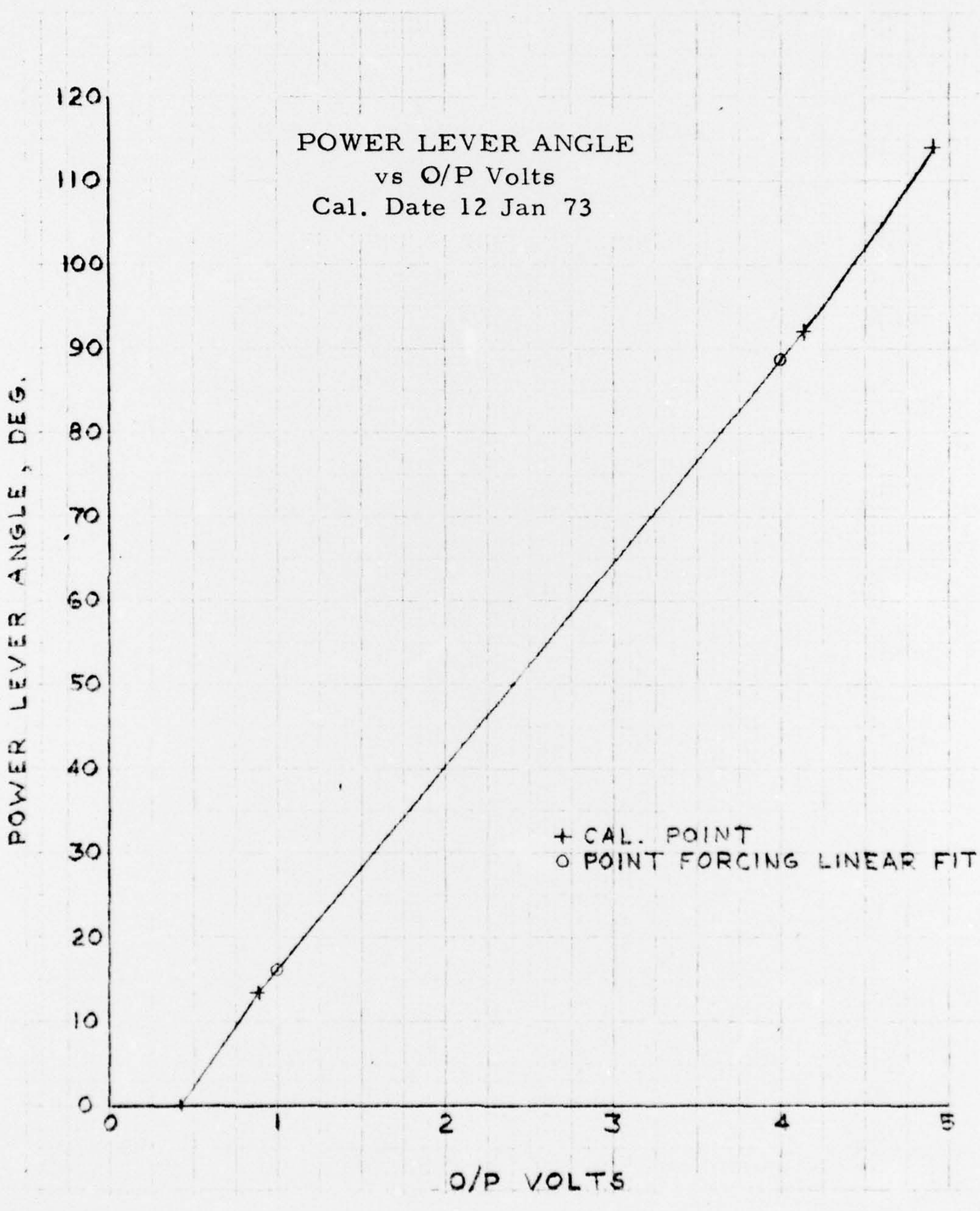
Indicated Airspeed

Main Fuel Flow Turbine

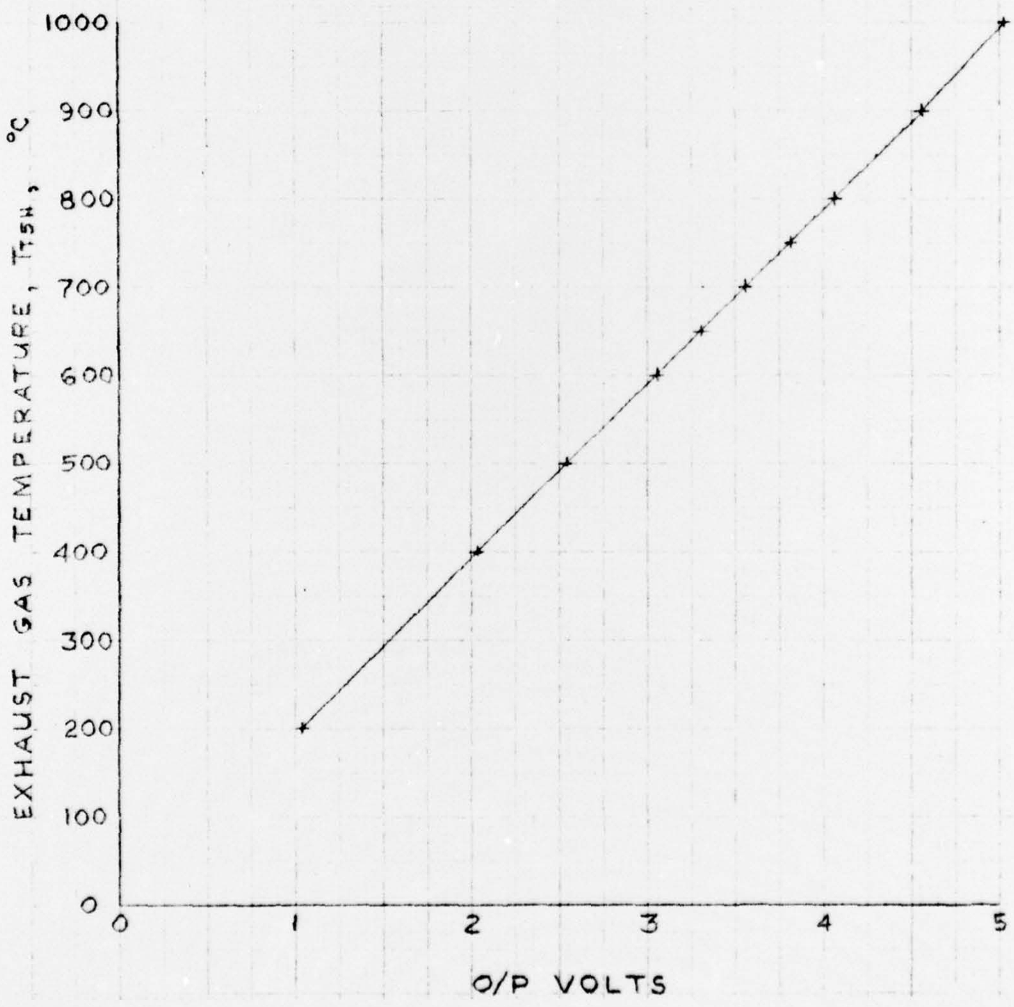
A/B Fuel Flow Turbine

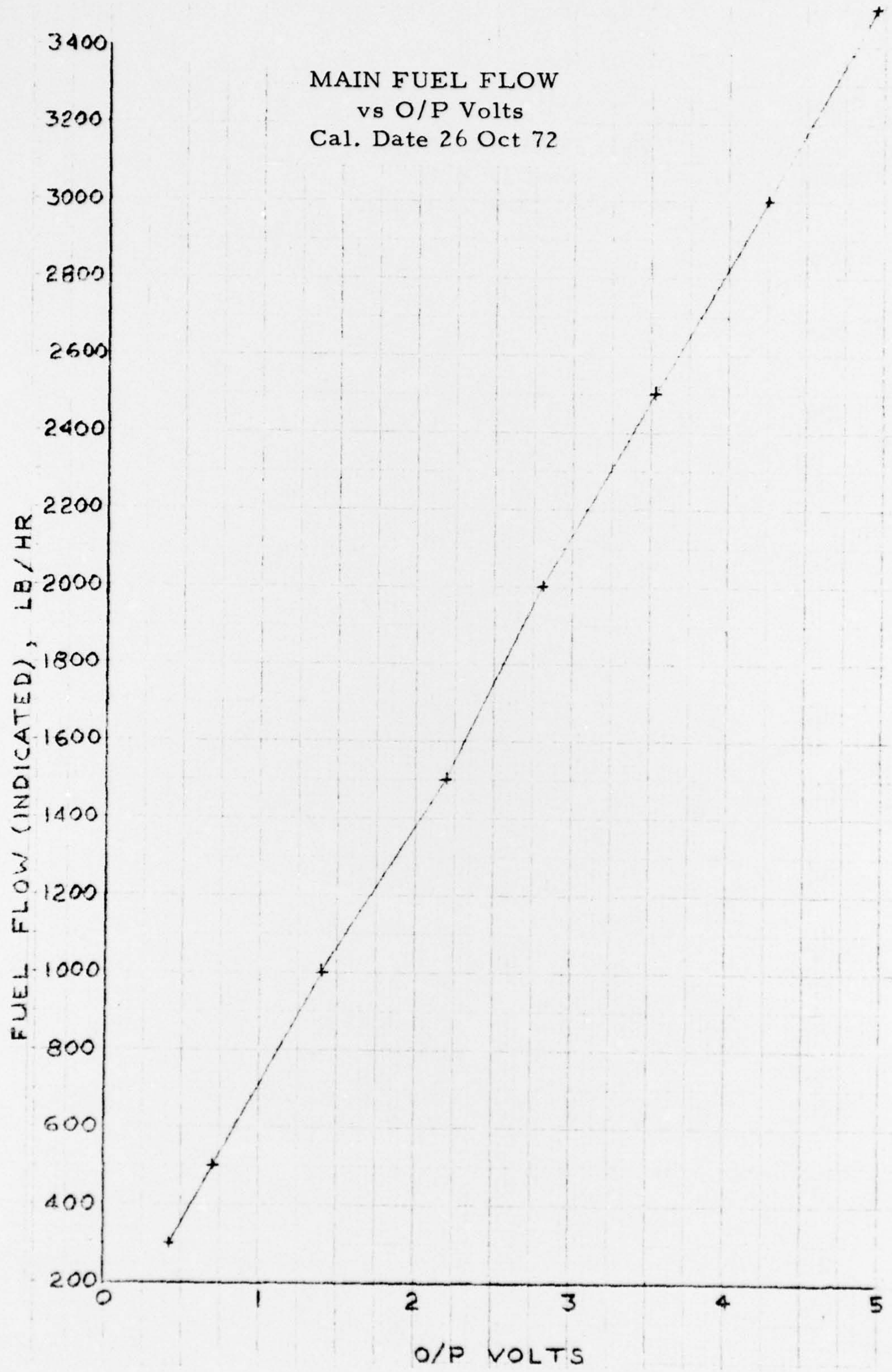


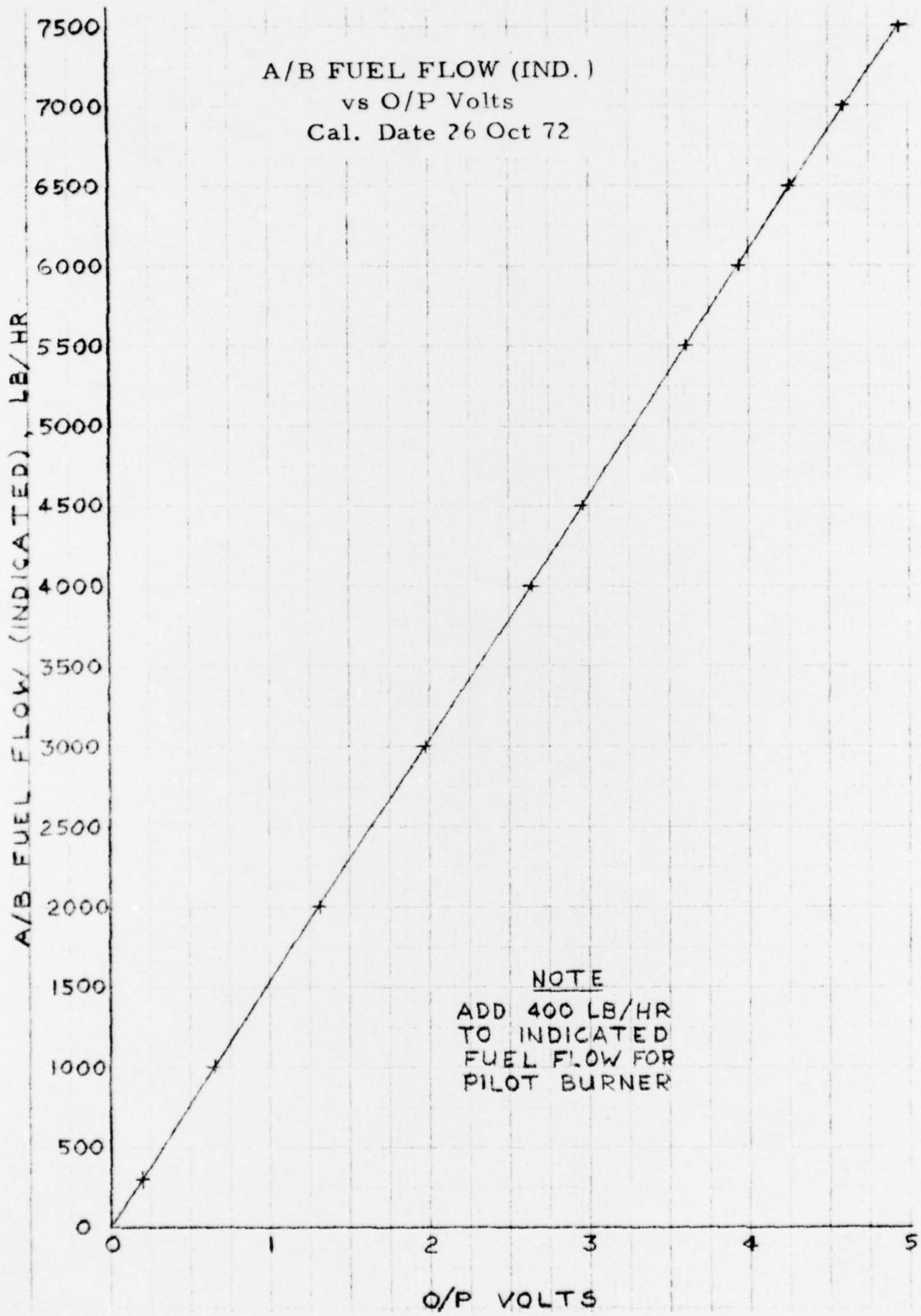
COMPRESSOR STATIC PRESSURE P_{s3}
vs O/P Volts
Cal. Date 26 Jan 73
PS3 (PSIA)
140
130
120
110
100
90
80
70
60
50
40
30
20
10
0
0
1
2
3
4
5
O/P VOLTS


POWER LEVER ANGLE
vs O/P Volts
Cal. Date 12 Jan 73
POWER LEVER ANGLE, DEG.
O/P VOLTS
+ CAL. POINT
o POINT FORCING LINEAR FIT
0
10
20
30
40
50
60
70
80
90
100
110
120
0
1
2
3
4
5

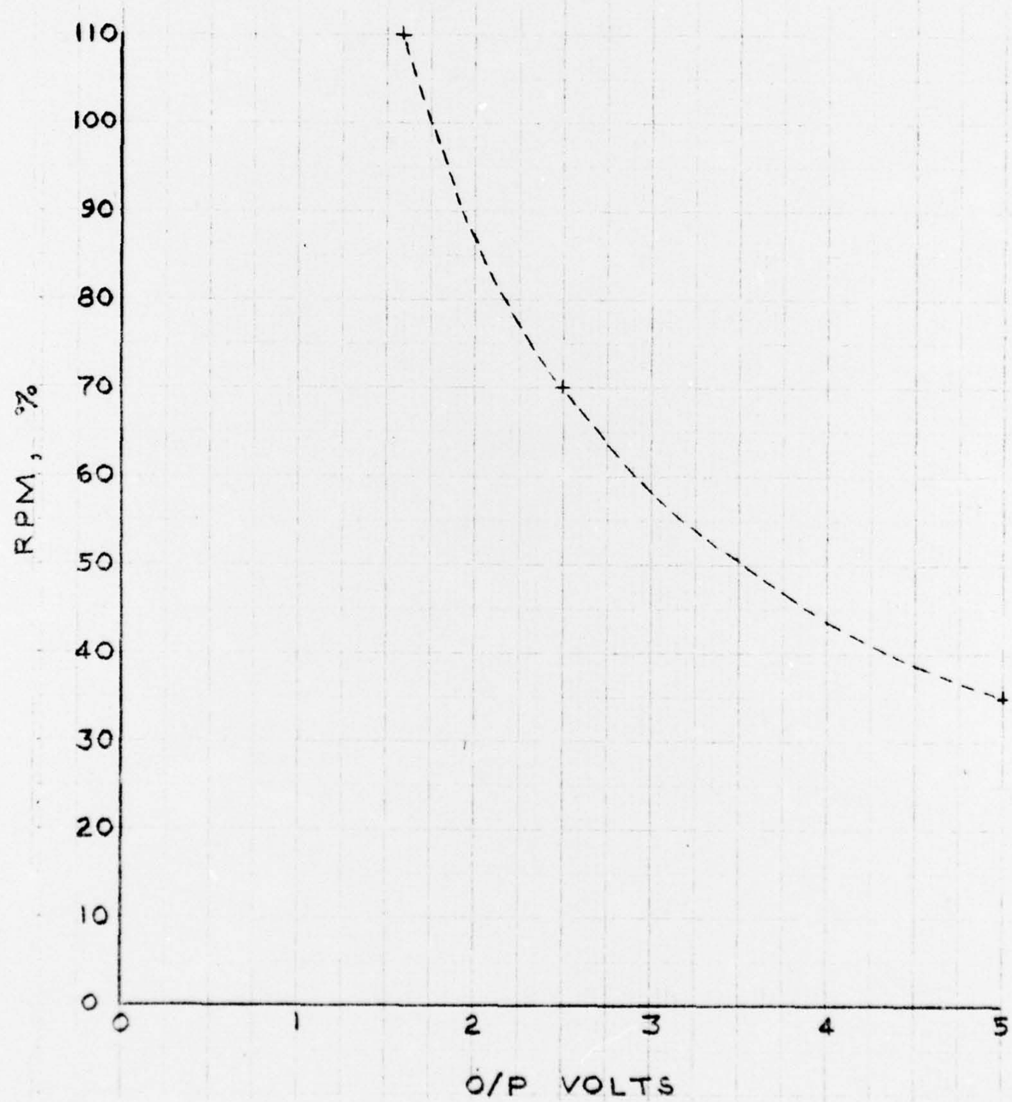
EXHAUST GAS TEMPERATURE
vs O/P Volts
Cal. Date 7 Nov 72

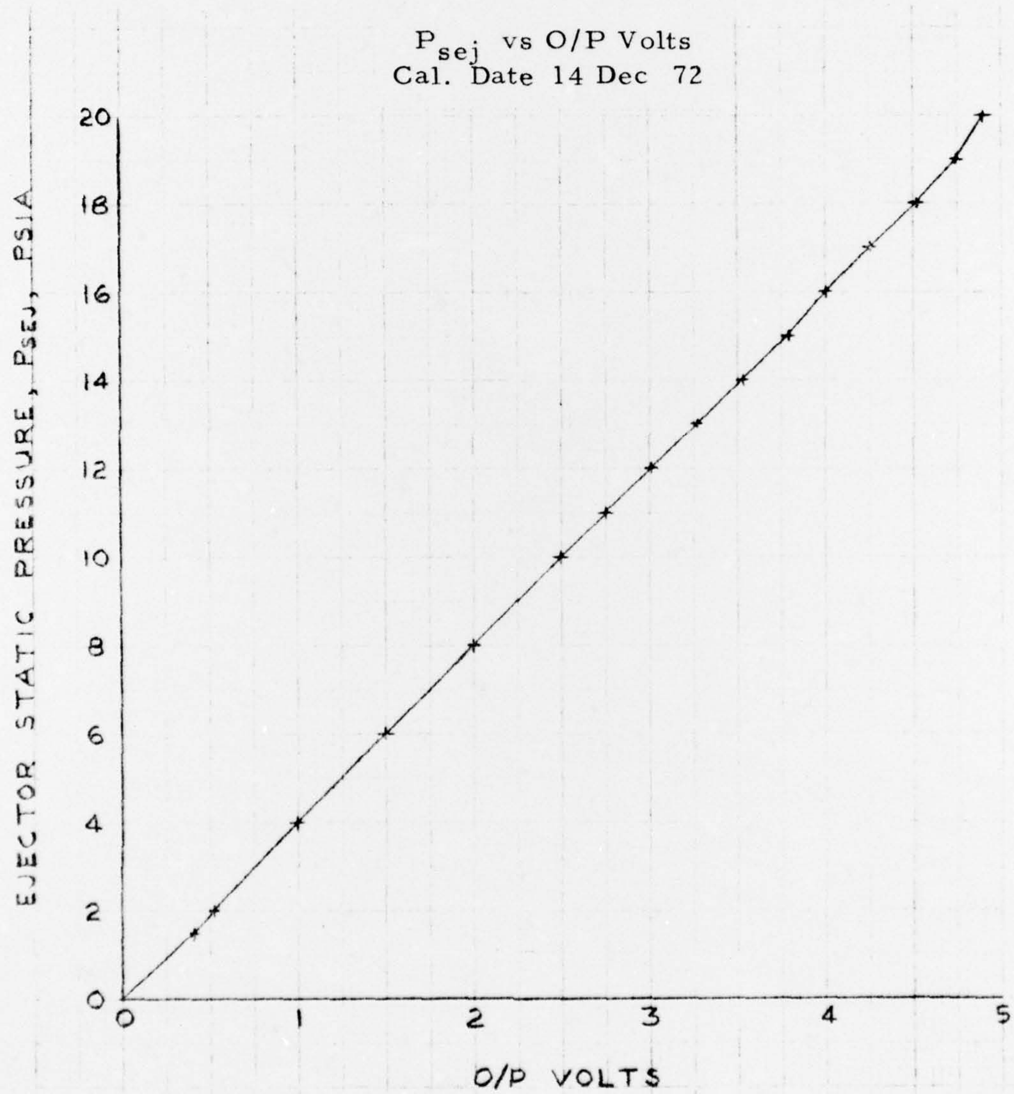



MAIN FUEL FLOW
vs O/P Volts
Cal. Date 26 Oct 72
FUEL FLOW (INDICATED), LB/HR
3400
3200
3000
2800
2600
2400
2200
2000
1800
1600
1400
1200
1000
800
600
400
200
0
1
2
3
4
5
O/P VOLTS


A/B FUEL FLOW (IND.)
vs O/P Volts
Cal. Date 26 Oct 72
A/B FUEL FLOW (INDICATED), LB/HR
7500
7000
6500
6000
5500
5000
4500
4000
3500
3000
2500
2000
1500
1000
500
0
0
1
2
3
4
5
O/P VOLTS
NOTE
ADD 400 LB/HR
TO INDICATED
FUEL FLOW FOR
PILOT BURNER

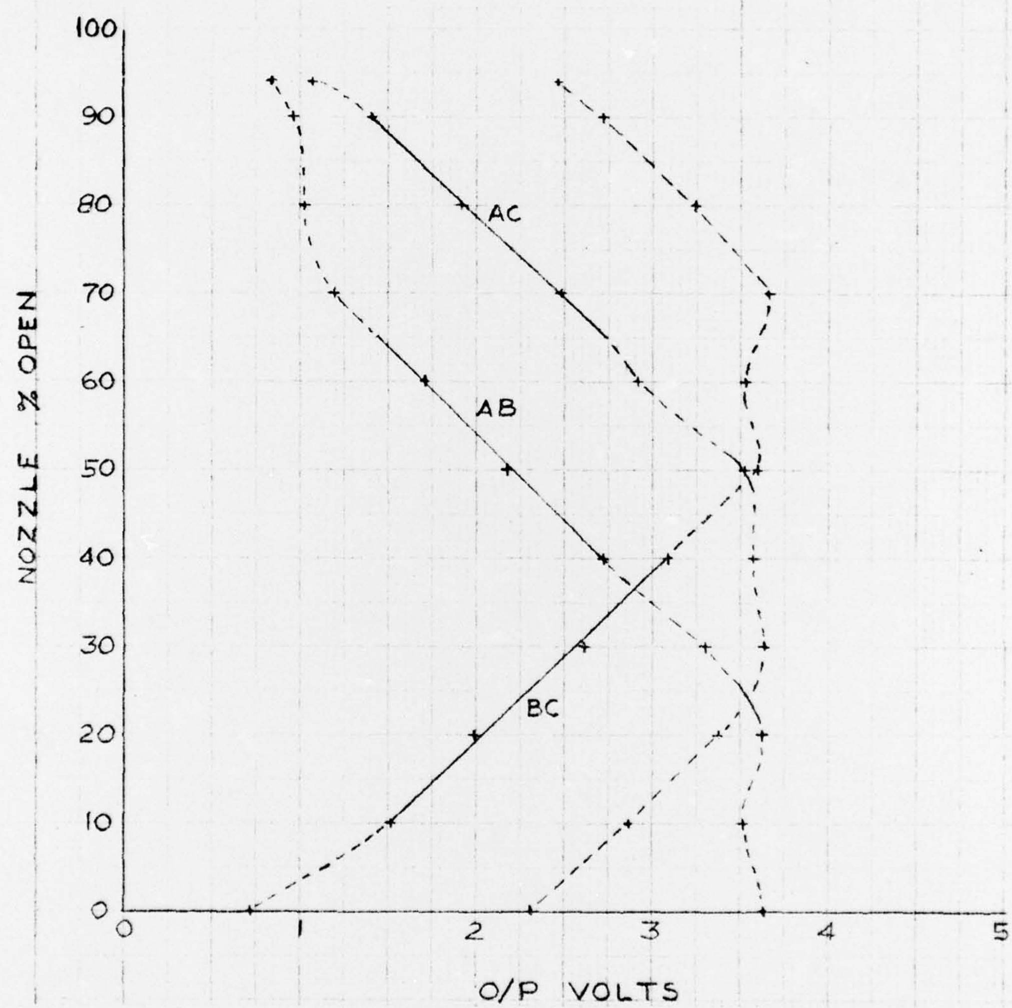
PER CENT RPM
vs O/P Volts
Cal. Date 7 Sep 72



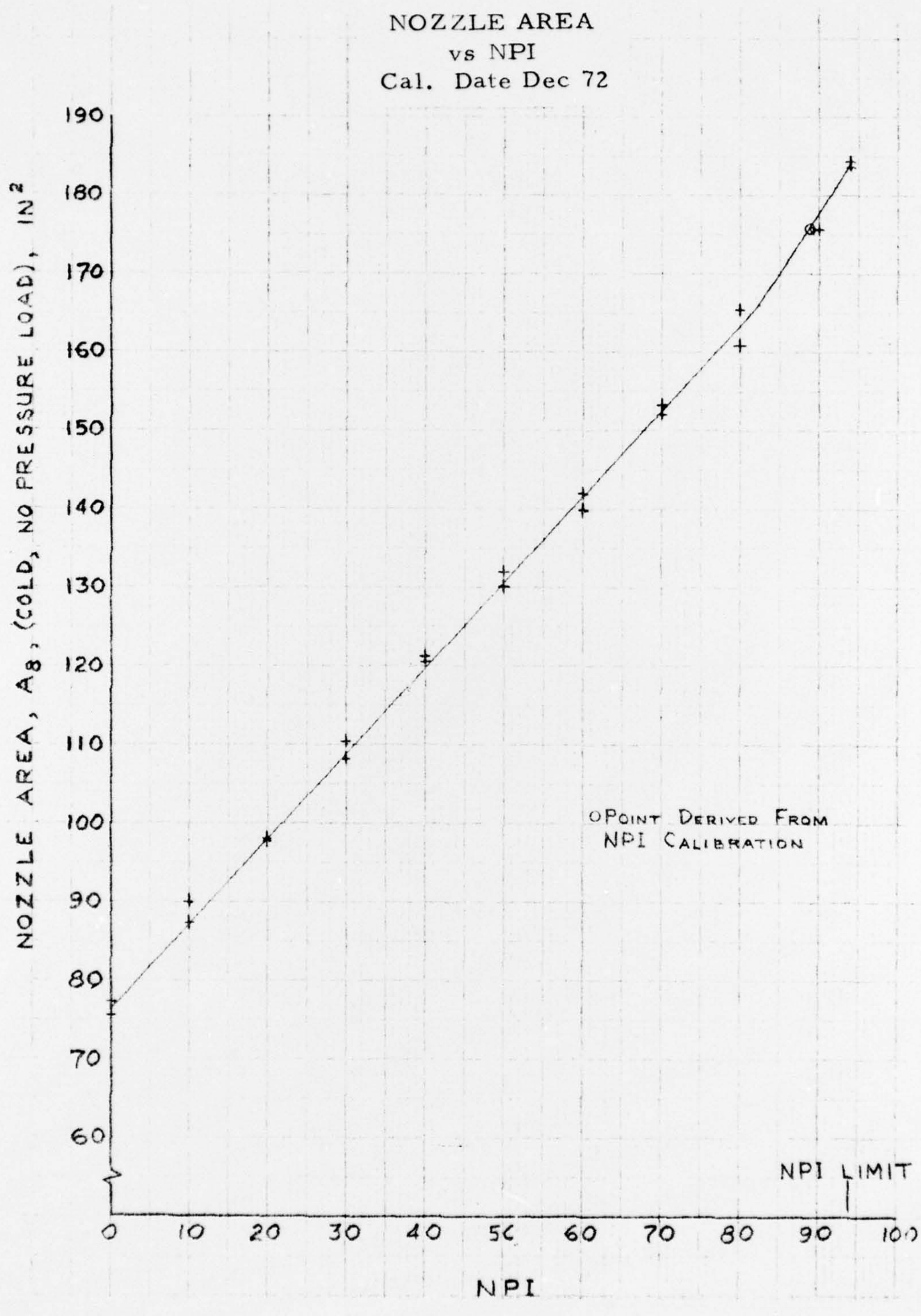


P_{sej} vs O/P Volts
Cal. Date 14 Dec 72
EJECTOR STATIC PRESSURE, P_{SEJ}, PSIA
O/P VOLTS
0
2
4
6
8
10
12
14
16
18
20
0
1
2
3
4
5

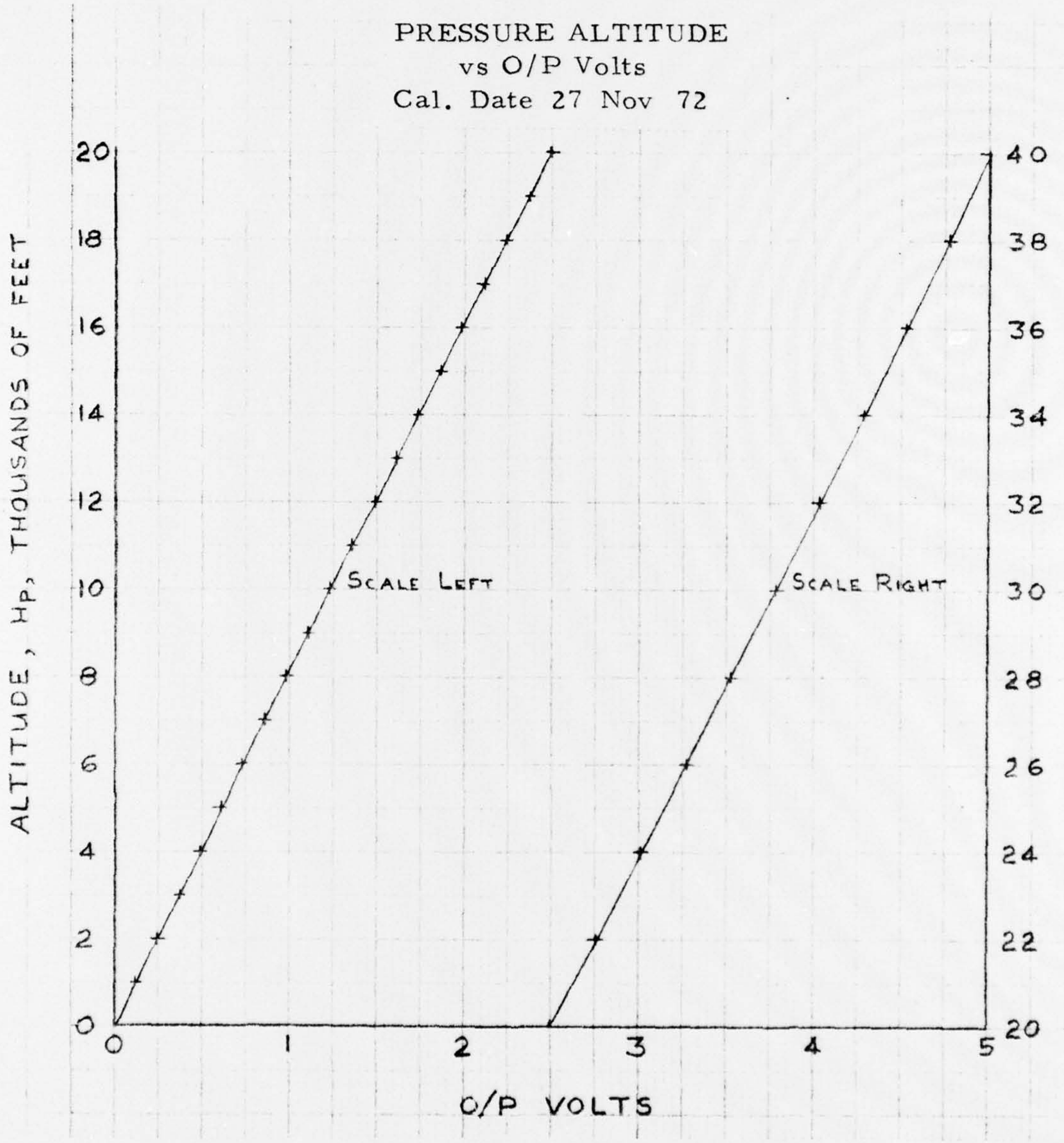
NOZZLE POSITION
vs O/P Voltage
Cal. Date 6 Feb 73

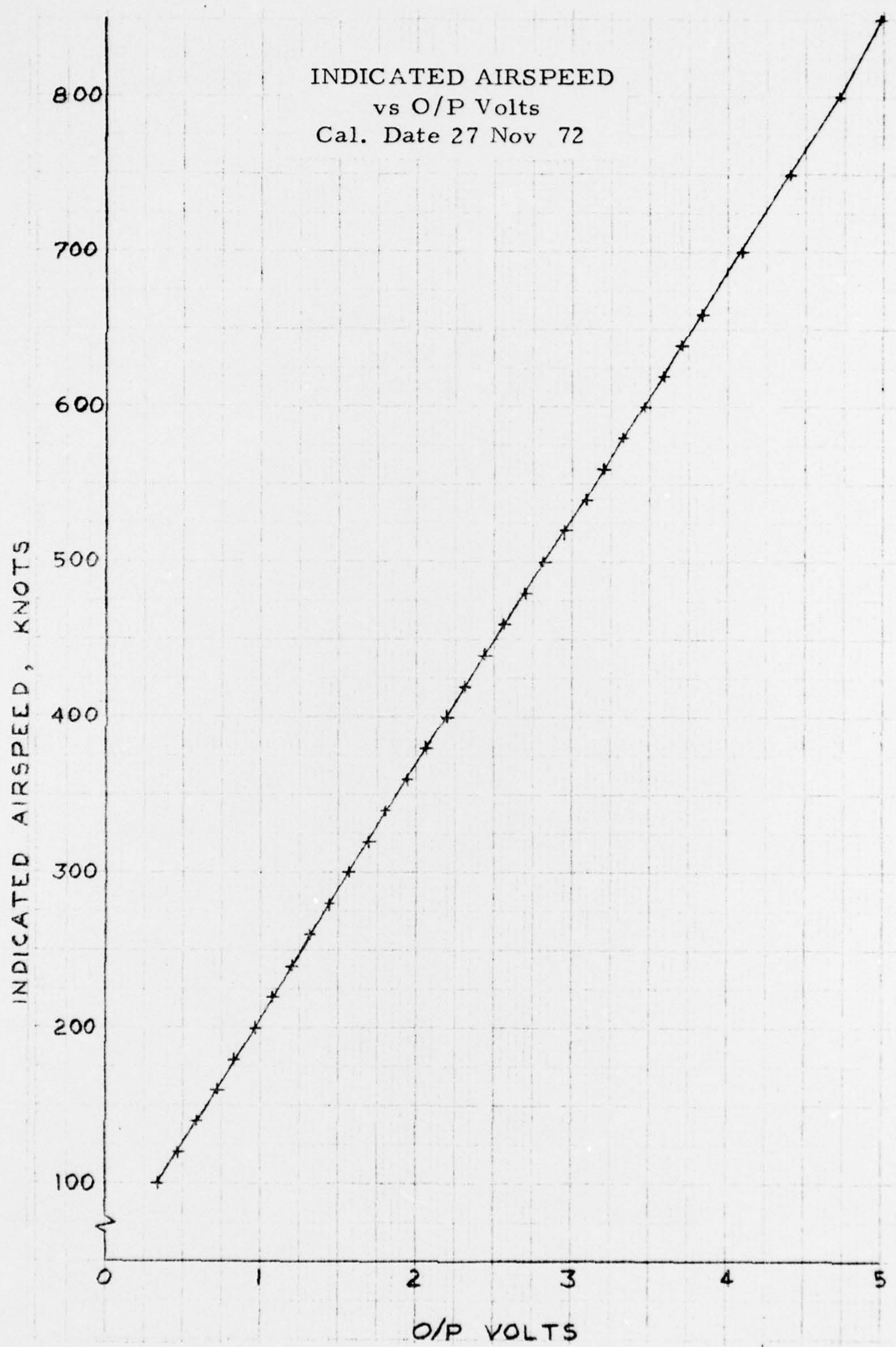


Rules:
If AB > 65 use AC
If AB ≤ 40 use BC
If $65 \leq$ AB $<$ use AB



NOZZLE AREA
vs NPI
Cal. Date Dec 72
NOZZLE AREA, A_8, (COLD, NO PRESSURE LOAD), IN^2
190
180
170
160
150
140
130
120
110
100
90
80
70
60
○ Point Derived From
NPI Calibration
NPI LIMIT
0
10
20
30
40
50
60
70
80
90
100
NPI


PRESSURE ALTITUDE
vs O/P Volts
Cal. Date 27 Nov 72
ALTITUDE, H_P, THOUSANDS OF FEET
O/P VOLTS
SCALE LEFT
SCALE RIGHT
0
2
4
6
8
10
12
14
16
18
20
22
24
26
28
30
32
34
36
38
40
0
1
2
3
4
5


INDICATED AIRSPEED
vs O/P Volts
Cal. Date 27 Nov 72
INDICATED AIRSPEED, KNOTS
800
700
600
500
400
300
200
100
0
1
2
3
4
5
O/P VOLTS

MAIN FUEL FLOW

[illegible]

	Flow Transmitter	[illegible]:	[illegible]
[illegible]:	[illegible] 1	S/N NO.:	[illegible]
R.T. NO.:	DSO-255297	DATE:	January 3rd, 1972

P.P.H.	HERTZ
[illegible]	51
[illegible]	83
[illegible]	167
[illegible]	[illegible]
[illegible]	[illegible]
[illegible]	676
[illegible]	[illegible]
[illegible]	1015
[illegible]	[illegible]
8000	[illegible]
9000	1523

[illegible] of indicated flow using Calibrating Fluid MIL-C-7024B Type II @ [illegible]°F S.G. 0.770 @ 60°F, 0.760 @ 80°F.

CALIBRATED BY:.......................

REVIEWED BY:.......................

Bendix

A/B FUEL FLOW

[illegible]

[illegible] [illegible]

[illegible]: 3/4-[illegible]01 SERIAL NO.: 3[illegible]

P.O. NO.: D30-259206 DATE: January 3rd, 1972

P.P.H.	HERTZ
300	49
600	82
1000	165
2000	331
3000	498
4000	663
5000	831
6000	999
7000	116[illegible]
8000	13[illegible]
9000	1496

Turbine is ±2% of indicated flow using Calibrating Fluid MIL-C-7024B Type II @ [illegible] S.G. 0.770 @ 60°F, 0.760 @ [illegible]°F.

CALIBRATED BY:............................

REVIEWED BY:............................

Bendix

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